Vol II Launch Vehicle
ILRS-NASA Final Report

Report H367
12 June - 1969

FOREWORD

As originally negotiated, this portion of the Integral Launch and Re-entry vehicle (ILRV) study was to cover the selection and detailed analysis of potential low cost booster candidates However, by direction of NASA Langley the latter portion of the study (detailed analysis) has been deleted This volume thus contains only the expendable launch vehicle selection portion of ~~the~~ ILRV Study It is submitted in partial fulfillment of NASA Contract No NAS9-9204 The study described here was conducted for the Langley Research Center under the direction of K. Edwards. Data documented herein was prepared by the Advanced Space and Launch Systems Directorate of the McDonnell Douglas Astronautics Company-Western Division under the direction of M. B Adams Other principal contributors are listed in the acknowledgements

The information presented here was compiled and prepared during the period March 1969 to July 1969.

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ABSTRACT

A low-cost space transportation system is a requisite to the future more economical space exploration A study was conducted to determine (1) the potential economics of a reusable transportation system and (2) the degree of reusability that would offer the most promising operational space transportation/logistics system The expendable portion of the study, as originally contracted, was directed toward the selection and detailed analysis of the best low-cost expendable launch vehicle that could be used with a reusable spacecraft. However, by direction of NASA Langley the latter portion of this analysis has been deleted The remaining portion of the study is the subject of this volume

Some 32 launch vehicle concepts with low-cost potential were sized to a minimum cost criterion and the five most promising were selected for further analysis. This volume presents the launch vehicles that were analyzed and the five most promising low-cost launch vehicle concepts All five concepts used solid propulsion in the lower stages The work presented in this volume was conducted by the Western Division of the McDonnell Douglas Astronautics Company.

CONTENTS

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I INTRODUCTION AND SUMMARY

The objective of the MDAC ILRV study was to determine what degree of reusability would offer the lowest-cost operational logistics systems Therefore, both reusable spacecraft with expendable launch vehicles and totally reusable launch vehicle/spacecraft (with expendable boost propellant tanks) were to be considered However, about halfway through the study, the expendable portion of the study was cancelled by a NASA redirection, and only the totally reusable approaches were to be considered The expendable launch vehicle work that had been conducted is reported in this volume

The expendable launch vehicle study was divided into two parts the first was devoted to determining the two to five most promising configurations that would offer the lowest cost, the second, which was cancelled, would have been devoted to a more detailed analysis of the two to five configurations to estimate more closely both the operational and developmental costs These costs would then be combined with the reusable spacecraft to determine total program costs

The first part of the study was completed and is the subject of this report Some 32 launch vehicle concepts made up of the following 6 propulsion concepts were sized to the criterion of minimum cost to help determine the most promising low-cost expendable system

1 Solid propulsion
2 Pressure-fed N_2O_4/UDMH propulsion
3 Pump-fed N_2O_4/UDMH propulsion
4 Pressure-fed LO_2/RP-1 propulsion
5 Pump-fed LO_2/RP-1 propulsion
6 Pump-fed LO_2/LH_2 propulsion

Each of the 32 vehicles were sized to provide minimum cost for a payload of 25,000 lb (82,500 lb thrown weight) to a 100 nmi Earth orbit After they were sized, the launched cost of each concept (excluding launch-site housekeeping expenses) were estimated All costs were based on a low-cost, operational aerospace quality operation This analysis indicated that the following five configurations would offer the best low-cost potential

1 Solid-propellant first and second stage with a LO_2/RP-1 third stage
2 Solid-propellant first and second stage with a LO_2/LH_2 third stage
3 Solid-propellant first stage with a LO_2/LH_2 second stage
4. Solid-propellant first stage with a LO_2/RP-1 second stage
5 Solid-propellant first stage with a LO_2/RP-1 second and third stage

These selections were the result of a systematic analysis that is outlined in Figure 1-1 A brief statement of each of the eight tasks in Figure 1-1 follows

1 1 Preliminary Screening (Section II) - If all permutations of the six propulsion concepts in two- and three-stage configurations were considered, there would be a 150 launch vehicle concepts to analyze This possible matrix was reduced to 32 concepts by always putting higher energy stages above (and never below) lower energy stages, and assuming that the trends of many possible configurations that were not analyzed could be identified by noting the cost trends of similar configurations analyzed

1 2 Propulsion System Definition (Section 3 1) - Nominal propulsion system descriptions and performance parameters were determined for both lower- and upper-stage applications for the six propulsion concepts These parameters, selected to provide low cost and reliability, were based on existing systems, past studies, and inputs from propulsion companies These data were used to help determine the weights, costs, and sizing of Blocks 4 1, 4 2, and 5 0

1 3 Electromechanical Systems Definition (Section 3 2) - A simplified electronics and control system that would receive guidance signals from the spacecraft was described for each configuration This subsystem was conceived to provide a safe, reliable operation with a minimum amount of telemetry and complexity It was based on existing quality equipment

1 4 Structure Systems Definition (Section 3 3) - A "peg point" design for each of the 32 candidates was prepared to provide basic launch vehicle descriptions These data were used to determine inert weights and costs as a function of the propellant weight of the various stages A major objective was to assure that all designs were based on common design criteria so the comparison would be valid

1 5 Parametric Weight Analyses (Section 4 1) - Inert weight of each of the stages to be used in the sizing analysis was determined as a function of stage size (propellant weight) A major objective was to assure that all weights were realistic and consistent These weights were estimated at the major subsystem level and then combined for total weight The subsystem weights and descriptions were used in the parametric cost determination

1 6 Parametric Cost Analyses (Section 4 2) - The cost of each stage as a function of size (propellant weight) was determined as an input to the launch vehicle sizing These data were also used as an input to the preliminary cost estimate for each vehicle after it had been sized These costs were estimated for

LOW COST EXPENDABLE LAUNCH VEHICLE STUDY

FUNCTIONAL FLOW DIAGRAM

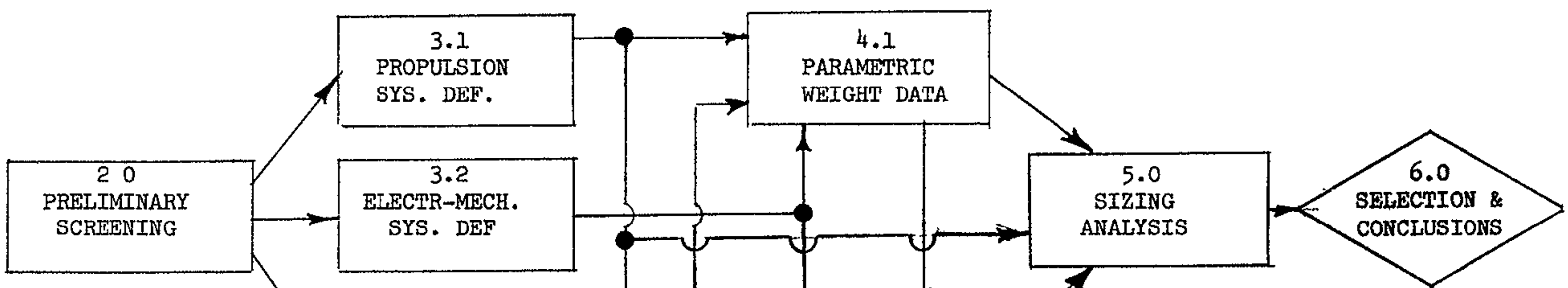


FIGURE 1-1

the first unit, were based on a low cost aerospace quality program, and included delivered hardware Launch and operational costs were not included but were added after the sizing had been completed for the selection task (See section VI)

1 7 Sizing Analysis (Section V) - Each of the 32 launch vehicle concepts was sized to place a lifting body spacecraft with a nominal cargo weight of 25,000 lb and 12 men (82,500 lb of thrown weight) into a low Earth orbit The size distribution of the stages for each candidate was determined to provide minimum cost

1 8 Launch Vehicle Selection for Further Analysis (Section VI) - After cost estimates that included manufacturing, propellant, and launch operations were made for the 32 concepts, the five that offered the lowest cost were selected

The remainder of this report is devoted to a detailed description of the tasks and presentation of the data generated

II PRELIMINARY LAUNCH VEHICLE SCREENING

Thirty-two launch vehicle concepts were selected from the potential 150 that would result if all six propulsion types were stacked in all possible two- and three-stage arrangements This reduction was made by first assuming that higher energy stages should always go above energy stages, then arranging a matrix so that trends of the many unanalyzed arrangements could be noted

Table 2-1 lists the 32 concepts and Tables 2-2 and 2-3 show them in matrix form The X's with subscripts indicate those vehicles that were investigated These vehicles were selected so that data for the remaining launch vehicle combinations in the matrix, which will not be cost sized, can be determined As an example, a pressure-fed N_2O_4/UDMH first stage with a pump-fed LO_2-RP second stage will not be cost sized The data required for this launch vehicle can be determined from the pressure-fed LO_2-RP first stage with a pump-fed LO_2-RP second stage, (Configuration X_{20}) by accounting for the difference between the pressure-fed N_2O_4/UDMH and the pressure-fed LO_2-RP first stages, which can be obtained by comparing Configurations X_{16} and X_{22} From inspecting Figure 5-2 both Configurations 16 and 22 have the same first unit cost versus gross weight relationships, therefore, the difference between a pressure-fed N_2O_4/UDMH and a pressure-fed LO_2-RP first stage is negligible Since LO_2-RP and N_2O_4/UDMH pressure-fed first stages have the same cost effectiveness, a pressure-fed N_2O_4/UDMH first stage with a pump-fed LO_2-RP second stage would have the same cost/weight characteristics as Configuration 20 The numerically subscripted 0's shown in Table 2-3 indicate those vehicles of the matrix which have both first and second stages the same, whereas the subscripted X's have similar second and third stages

Figure 2-1 was prepared to illustrate the principle that a high energy stage, even with the high performance of LO_2/LH_2, should not be placed below a lower energy stage This figure shows relative cost versus the gross weight of a two stage LO_2/RP-1 + LO_2/LH_2 vehicle The cost of a vehicle with LO_2/RP-1 as a first stage is less than one-half the cost of a vehicle with LO_2/LH_2 as the first stage The data on this curve is for a thrown weight of 80, 000 lb into a low Earth orbit The cost of the LO_2/LH_2 stage is about four times that of the LO_2/RP-1 stage The total cost relationship shown is due to the obvious fact that lower stages must always be substantially larger than upper stages because they are pushing more total weight

Table 2-1

CANDIDATE EXPENDABLE LAUNCH VEHICLES

Conf No	1st Stage	2nd Stage	3rd Stage
X1	Solid rocket motor	Solid rocket motor	---
X2	Solid rocket motor	Solid rocket motor	Solid rocket motor
X3	Solid rocket motor	Solid rocket motor	LO_2-RP, pump-fed
X4	Solid rocket motor	Solid rocket motor	LO_2-LH_2, pump-fed
X5	Solid rocket motor	Storable, pressure-fed	---
X6	Solid rocket motor	LO_2-RP, pressure-fed	---
X7	Solid rocket motor	Storable, pump-fed	---
X8	Solid rocket motor	Storable, pump-fed	Storable, pump-fed
X9	Solid rocket motor	LO_2-RP, pump-fed	---
X10	Solid rocket motor	LO_2-RP, pump-fed	LO_2-RP, pump-fed
X11	Solid rocket motor	LO_2-LH_2, pump-fed	---
X12	Storable, pressure-fed	Storable, pressure-fed	---
X13	Storable, pressure-fed	Storable, pressure-fed	Storable, pressure-fed
X14	Storable, pressure-fed	Storable, pump-fed	---
X15	Storable, pressure-fed	LO_2-RP, pump-fed	LO_2-RP, pump-fed
X16	Storable, pressure-fed	LO_2-LH_2, pump-fed	---
X17	LO_2-RP, pressure-fed	LO_2-RP, pressure-fed	---
X18	LO_2-RP, pressure-fed	LO_2-RP, pressure-fed	LO_2-RP, pressure-fed
X19	LO_2-RP, pressure-fed	Storable, pump-fed	Storable, pump-fed
X20	LO_2-RP, pressure-fed	LO_2-RP, pump-fed	---
X21	LO_2-RP, pressure-fed	LO_2-RP, pump-fed	LO_2-RP, pump-fed
X22	LO_2-RP, pressure-fed	LO_2-LH_2, pump-fed	---
X23	Storable, pump-fed	Storable, pump-fed	---
X24	Storable, pump-fed	Storable, pump-fed	Storable, pump-fed
X25	Storable, pump-fed	LO_2-RP, pump-fed	LO_2-RP, pump-fed
X26	Storable, pump-fed	LO_2-LH_2, pump-fed	---
X27	LO_2-RP, pump-fed	LO_2-RP, pump-fed	--
X28	LO_2-RP, pump-fed	LO_2-RP, pump-fed	LO_2-RP, pump-fed
X29	LO_2-RP, pump-fed	LO_2-RP, pump-fed	LO_2-LH_2, pump-fed
X30	LO_2-RP, pump-fed	LO_2-LH_2, pump-fed	---
X31	LO_2-LH_2, pump-fed	LO_2-RP, pump-fed	---
X32	LO_2-LH_2, pump-fed	LO_2-LH_2, pump-fed	---

Table 2-2

EXPENDABLE LAUNCH VEHICLES TWO STAGE CANDIDATES

First Stages	Second Stages					
		Liquids (Pressure-Fed)		Liquids (Pump-Fed)		
	Solid	N_2O_4/UDMH	LO_2-RP	N_2O_4/UDMH	LO_2-RP	LO_2-LH_2
1 Solid	X_1	X_5	X_6	X_7	X_9	X_{11}
2 Liquids (Pressure Fed)						
N_2O_4/UDMH		X_{12}		X_{14}		X_{17}
LO_2-RP			X_{17}		X_{20}	X_{22}
3 Liquids (Pump Fed)						
N_2O_4/UDMH				X_{23}		X_{26}
LO_2-RP					X_{27}	X_{30}
LO_2-LH_2					X_{31}	X_{32}

X = Second and third stages the same

O = First and second stages the same

Table 2-3

EXPENDABLE LAUNCH VEHICLES THREE STAGE CANDIDATES

First Stages	Second Stages and Third Stages					
		Liquid (Pressure-Fed)		Liquids (Pump-Fed)		
	Solids	N_2O_4/UDMH	LO_2-RP	N_2O_4/UDMH	LO_2-RP	LO_2-LH_2
1 Solid	X_2			X_8	X_{10} O_3	O_{34}
2 Liquids (Pressure Fed)						
N_2O_4/UDMH		X_{13}			X_{15}	
LO_2-RP			X_{18}	X_{19}	X_{21}	
3 Liquids (Pump-Fed)						
N_2O_4/UDMH				X_{24}	X_{25}	
LO_2-RP					X_{28}	O_{29}

X = Second and third stages the same

O = First and second stages the same

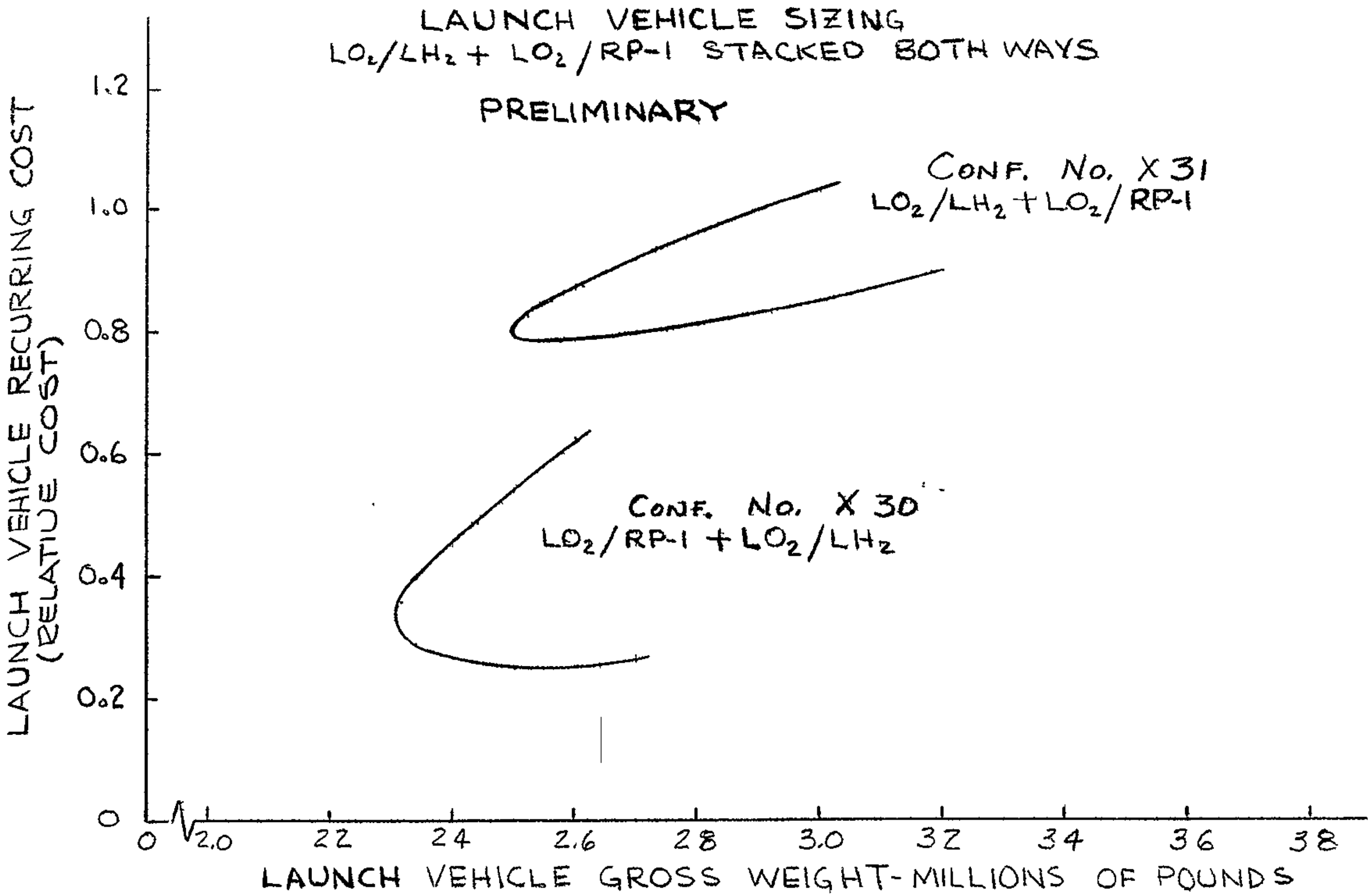


Figure 2-1 Two Stage LH_2/RP-1 Launch Vehicles

In order to further reduce the number of candidates, it was assumed that the upper stages of all three stage candidates would utilize the same propellant/engine concept Exceptions to this are candidates 03 and 04 both of which use 260 in diameter solid rocket motors as first and second stages These candidates were added later in the study when it became apparent that solid rocket motor stages would produce the more attractive candidates Another assumption, which was to have been substantiated during the second phase of the study, was that the 260 in diameter solid was representative of the solid motor concept An investigation of clusters of 156 in diameter SRM's in lieu of the single 260 in SRM was to have been a part of the detailed analysis planned for the second phase of the study

The numbers that are shown beside the concepts on Table 2 1 are used on occasion in this report Where this occurs this table can be a convenient reference

III SYSTEM DEFINITION

In order to develop realistic weight, cost, and performance data, the three major subsystems of each stage concept were defined at the subsystem level These subsystems are discussed under the three major vehicle categories Propulsion (Section 3 1), Astrionics (Section 3 2) and Structures (Section 3 3)

3 1 Propulsion System Definition - The objective of this portion of the study was to provide propulsion system data for the candidate liquid- and solid-propellant expendable launch vehicles These data included definition of the liquid engine and solid motor performance parameters, generation of parametric propulsion subsystem weights, and description of the various subsystems in sufficient depth to enable a realistic assessment of the subsystem recurring costs The governing subsystem selection criterion was minimum recurring cost, with minimum weight as a secondary criterion Following is a discussion of the work which was performed during the study

3 1 1 Liquid-Propulsion Systems - Both pump-fed and pressure-fed propulsion systems were evaluated during the study The propellant combinations used for the launch vehicles were LO_2/RP-1, N_2O_4/UDMH and LO_2/LH_2 The selection of LO_2/RP-1 was made because of its low cost, good performance, and high-bulk density In addition, the extensive experience with LO_2/RP-1 systems should favor low-cost operation Of the storable propellants, N_2O_4/UDMH offers reasonably high performance, and the use of UDMH results in low propellant costs relative to a blend of hydrazine and UDMH Oxygen/hydrogen was selected as being representative of high energy propellant at modest cost However, because of the hydrogen, the bulk density of the LO_2/LH_2 combinations is low (about 20 lb/cu ft) and consequently this combination was not considered for the pressure-fed launch vehicle application The high tank pressures associated with pressure-fed engines coupled with the low-bulk density of LO_2/LH_2 results in excessive tankage and pressurization system weights which would adversely affect the overall performance, hence cost-effectiveness, of this type of system Throughout this portion of the study, no attempt was made to optimize the subsystem parameters nor to conduct subsystem tradeoff studies prior to selection The large number of candidate launch vehicle systems and the broad nature of the total study precluded subsystem optimization Therefore, the propulsion operating parameters and subsystem selection were based on previous experience derived from related hardware programs and studies

3 1 1 1 Liquid Engine Parameters - Pump-Fed Candidates - The thermochemical performance for LO_2/RP-1 was computed for a range of chamber pressures, expansion ratios, and mixture ratios A survey of existing LO_2/RP-1 engines was made to determine typical operating engine parameters and specific impulse efficiencies which could be applied to the theoretical values For example, the F-1 engine operates at a mixture ratio of 2 27 chamber pressure of 1, 100 psia and nozzle area ratio of 16 1 The parameters of this engine were selected as being near representative for the booster engines The specific impulse efficiency of the F-1 engine is about 89% of shifting equilibrium (SE) An efficiency of 88% was used to represent the low-cost approach, since a low-cost F-1 would lose about 2 to 3 sec of specific impulse This reduction was based on a discussion with Rocketdyne representatives

The only existing upper stage LO_2/RP-1 engine is the Atlas sustainer engine This engine operates at a mixture ratio of 2 27, chamber pressure of 706 psia, and nozzle expansion ratio of 25 An overall specific impulse efficiency of about 91% was calculated for this engine Again, assuming that the low-cost approach would result in a degradation in efficiency, an efficiency of 90% was selected for the upper stage engines A chamber pressure of 600 psia was selected for the upper stage LO_2/RP-1 engines (The difference in vacuum specific impulse for LO_2/RP-1 is less than one second for chamber pressures between 600 and 1, 000 psia) Nozzle expansion ratios of 40 and 60 were selected for the second and third stages, respectively These values are in keeping with current practice

Both Aerojet and Rocketdyne are studying low cost storable, pump-fed booster engines The engine mixture ratio selected by these manufacturers was 2 6 for the N_2O_4/UDMH combination Quoted performance at a chamber pressure of 1, 000 psia and an area ratio of 15 was about 89% of theoretical SE This value was therefore selected for the study

The storable upper stage engine parameters were kept the same as for the LO_2/RP-1 engines A chamber pressure of 600 psia was selected for the second and third stages with a nozzle expansion ratio of 40 and 60, respectively The vacuum specific impulse efficiency was increased by 1% over the sea-level value The reason for this slight increase in efficiency is due to the effects of the gas generator turbine exhaust which contribute an increased percentage of the total thrust at vacuum

Delivered performance for the LO_2/LH_2 combination is based on a review of three existing engines J-2, M-1, and RL-10 The M-1 was redesigned for first stage use with an area ratio of 18 1 and delivered 93 2% of theoretical specific impulse The J-2 and RL-10, upper-stage engines, deliver 95 0% and 96 0%, respectively (The specific impulse efficiency is higher for the RL-10 engine because it uses an expander cycle rather than a gas generator (GG) cycle to drive the turbopumps) The performance of the low-cost engines was based on using a gas generator cycle and the engines were assumed to deliver 1% less efficiency than the M-1 and J-2 engines (GG cycles) The same engine chamber pressures and nozzle expansion ratios were assigned to the LO_2/LH_2 stages as for the LO_2/RP-1 and N_2O_4/UDMH stages The high chamber pressure engine currently being developed by Pratt and Whitney under contract to the Air Force was not considered for the expendable launch vehicle portion of this study This type of engine is not being developed under the low-cost design philosophy and the use of high chamber pressure should result in only a second-order effect on the overall results of this study

Table 3-1 summarizes the design parameters which were selected for the pump-fed engines In all cases, a pump inlet pressure of 50 psia was assumed for both the fuel and oxidizer tanks Tank design pressures were based on this pressure level without taking into account feed line losses and liquid head pressure These latter two effects were neglected because of the parametric nature of the study and should not affect the relative overall study results

The chamber pressures and expansion ratios chosen for LO_2/RP-1 and N_2O_4/UDMH pressure-fed candidates are consistent with present work in the low-cost area The first-stage chamber pressure is 300 psia, the lowest practical level The corresponding area ratio is 6 5 1 chosen to avoid nozzle flow separation at sea level The second stage operates at 300 psia in order to obtain a reasonable expansion ratio of 15 1 (considering geometry) The third-stage chamber pressure is reduced to 150 psia which, for the reduced thrust level, results in an expansion ratio of 25:1

The pressure-fed propellant performance was derived from theoretical shifting equilibrium calculations The assumed efficiency was 1% higher than the corresponding pump-fed systems which have slightly lower specific impulse efficiency due to energy extracted for pumping Selected parameters for the pressure-fed engines are summarized in Table 3-2 Tank pressures were assumed to be 150 psia greater than the chamber pressures

Table 3-1

LOW COST, PUMP-FED ENGINE PARAMETERS

Propellants	LO_2/RP-1	N_2O_4/UDMH	LO_2/LH_2
Mixture Ratio	2 27	2 6	5 5
First Stage			
• Chamber pressure	1,000 psia	1,000 psia	1,000 psia
• Area ratio	15	15	15
• Specific impulse (S L)	262 sec	253 sec	349 sec
• Specific impulse (vac)	298 sec	287 sec	401 sec
• I_s efficiency	88%	89%	92%
• Inlet pressures (both pumps)	50 psia	50 psia	50 psia
Second Stage			
• Chamber pressure	600 psia	600 psia	600 psia
• Area ratio	40	40	40
• Specific impulse (vac)	322 sec	301 sec	427 sec
• I_s efficiency	90%	90%	94%
• Inlet pressures (both pumps)	50 psia	50 psia	50 psia
Third Stage			
• Chamber pressure	600 psia	600 psia	600 psia
• Area ratio	60	60	60
• Specific impulse (vac)	326 sec	306 sec	433 sec
• I_s efficiency	90%	90%	94%
• Inlet pressures (both pumps)	50 psia	50 psia	50 psia

3 1 1 2 Parametric Engine Weights - Several engine manufacturers active in the low-cost area were consulted to determine the effect of low-cost design philosophy on engine weight Both Rocketdyne and Aerojet performed studies for Aerospace to define the characteristics of a low cost N_2O_4/UDMH pump-fed engine Based on their results, the conclusion can be made that while the overall performance of a low-cost engine is lower than for current high-performance (higher cost) storable propellant engines, the weight is about the same This may be contrary to trend that one would intuitively expect However the primary reason that the low-cost and high-cost engine is expected to weigh about the same is in the turbopump design approach Rather than run the pumps and turbines at different optimum speeds which necessitate gearboxes or separate turbopump assemblies, a substantial cost savings may be realized by running the turbines and pumps from a common

Table 3-2

LOW COST, PRESSURE-FED ENGINE PARAMETERS

Propellants	LO_2/RP-1	N_2O_4/UDMH
Mixture Ratio	2 27	2 6
First Stage		
• Chamber pressure	300 psia	300 psia
• Area ratio	6 5	6 5
• Specific impulse (S L)	229 sec	220 sec
• Specific impulse (vac)	281 sec	269 sec
• I_s efficiency	89%	90%
• Tank pressure (both tanks)	450 psia	450 psia
Second Stage		
• Chamber pressure	300 psia	300 psia
• Area ratio	15	15
• Specific impulse (vac)	306 sec	291 sec
• I_s efficiency	91%	91%
• Tank pressure (both tanks)	450 psia	450 psia
Third Stage		
• Chamber pressure	150 psia	150 psia
• Area ratio	25	25
• Specific impulse (vac)	313 sec	300 sec
• I_s efficiency	91%	91%
• Tank pressure (both tanks)	300 psia	300 psia

shaft at the same speed Additional cost savings may also be realized by using a low-speed, single-stage, low-efficiency turbine to drive the pumps rather than high-speed, multistage, high-efficiency turbines The low-efficiency, common shaft turbopump approach reduces cost and engine weight by elimination of gearbox assemblies This weight reduction is offset however by increases in nozzle and injector weight due to the low-cost design approach The net result is that the low-cost approach to pump-fed engine design results in slightly lower performance than for higher-cost engines while the overall weight of the low-cost engine is comparable to the higher-cost engine

A similar trend between the weight of a low-cost and high-cost LO_2/RP-1 is anticipated because of the similar density of these propellants to N_2O_4/UDMH Therefore, the parametric weights for the N_2O_4/UDMH and LO_2/RP-1 pump-fed engines were based on a weight estimating relationship which was derived from current high-performance engines This relationship is discussed in detail in Section IV and when checked against current and proposed engines were found to provide excellant correlation (See Equation 4-1, Section IV)

Because of the large density differences between LH_2 and LO_2, the use of a common shaft to drive the turbopumps is questionable Therefore, this approach towards low-cost LO_2/LH_2 engine design may not be practical and separate or gear-driven turbopumps may still be required as is current practice Therefore, the parametric engine weights for low-cost LO_2/LH_2 pump-fed engines were increased by 10% over the weight estimating relationship for existing LO_2/LH_2 engines This 10% increase was based on the results of discussions with several engine manufacturers

Both TRW and Rocketdyne who are active in the low-cost pressure-fed engine area were contacted to provide parametric weight data In addition, weight estimating relationships for pressure-fed engine weights were derived by MDAC In comparing the engine manufacturers data with the weight estimating relationships considerable disparity was noted This is not surprising in view of the limited thrust levels for which pressure-fed engines have actually been fabricated The data provided by the engine manufacturers did not agree and were considerably different at the higher thrust levels The weight estimating relationships derived by MDAC resulted in data between the limits of the two engine manufacturers Rather than decide between the data provided by the engine manufacturers, certain of their relationships were used in the MDAC derivations for pressure-fed engine weights This compromise approach should provide valid trends in estimating the engine weights over the thrust range covered in the study Following are the pressure-fed engine weight estimating relationships which were used in this study

$$\text{Weight of engine} = W_{EC} + W_{EXT} + W_{CH} + W_{INJ}$$

$$W_{EC} = \text{Weight of exit cone} = 0\ 289\ (\epsilon - 1)\ \frac{F}{C_f P_c}$$

(from $\epsilon = 1$ to $\epsilon = 10$)

$$W_{EXT} = \text{Weight of cone extension} = 0.48\,\epsilon^{(.3)}\frac{F}{C_f P_c}$$

(from $\epsilon = 10$ to $\epsilon = \epsilon$)

$$W_{CH} = \text{Weight of chamber} = 3 \times 10^{-5}\,\frac{4F}{C_F}\left(0.88\sqrt{\frac{4F}{C_f P_c \pi}} + 20\right) + 0.393\sqrt{\frac{4F}{C_f P_c \pi}}\left(0.88\sqrt{\frac{4F}{C_f P_c \pi}} + 20\right)$$

$$W_{INJ} = \text{Weight of injector} = (W_{EC} + W_{EXT} + W_{CH})\left(\frac{0.206 - 0.00271\epsilon}{0.794 + 0.00271\epsilon}\right)$$

The derived equations are based on a constant material thickness for chamber and exit cone up to an area ratio of 10:1, after which a radiation extension is assumed. The chamber L* was assumed to increase with thrust level. Injector and valve weights are based on the engine manufacturer's estimate of the percentage weight of these items to the total engine weight. The equations have been checked from thrust levels of 7,000 lb to 2,000,000 lb, with good agreement with actual hardware data at the low end.

3.1.1.3 Engine Configuration - At the outset of the program, it was decided that the consideration of single-engine stages would simplify the study effort and still provide the necessary weight and performance data to permit adequate cost effective evaluation comparison.

Single-engine configurations offer certain advantages over multiengine clusters. For example, the nozzle expansion ratio, hence specific impulse, is highest for a single engine because the nozzle exit area utilizes the maximum stage base area available. Two-engine stage configurations result in the lowest nozzle expansion ratio. As the number of engines are increased, the total available nozzle exit area begins to approach the single-engine configuration due to the increased efficiency of base area utilization. This effect upon maximum available nozzle expansion ratio is shown in Figure 3-1. Additionally, the single-engine simplifies

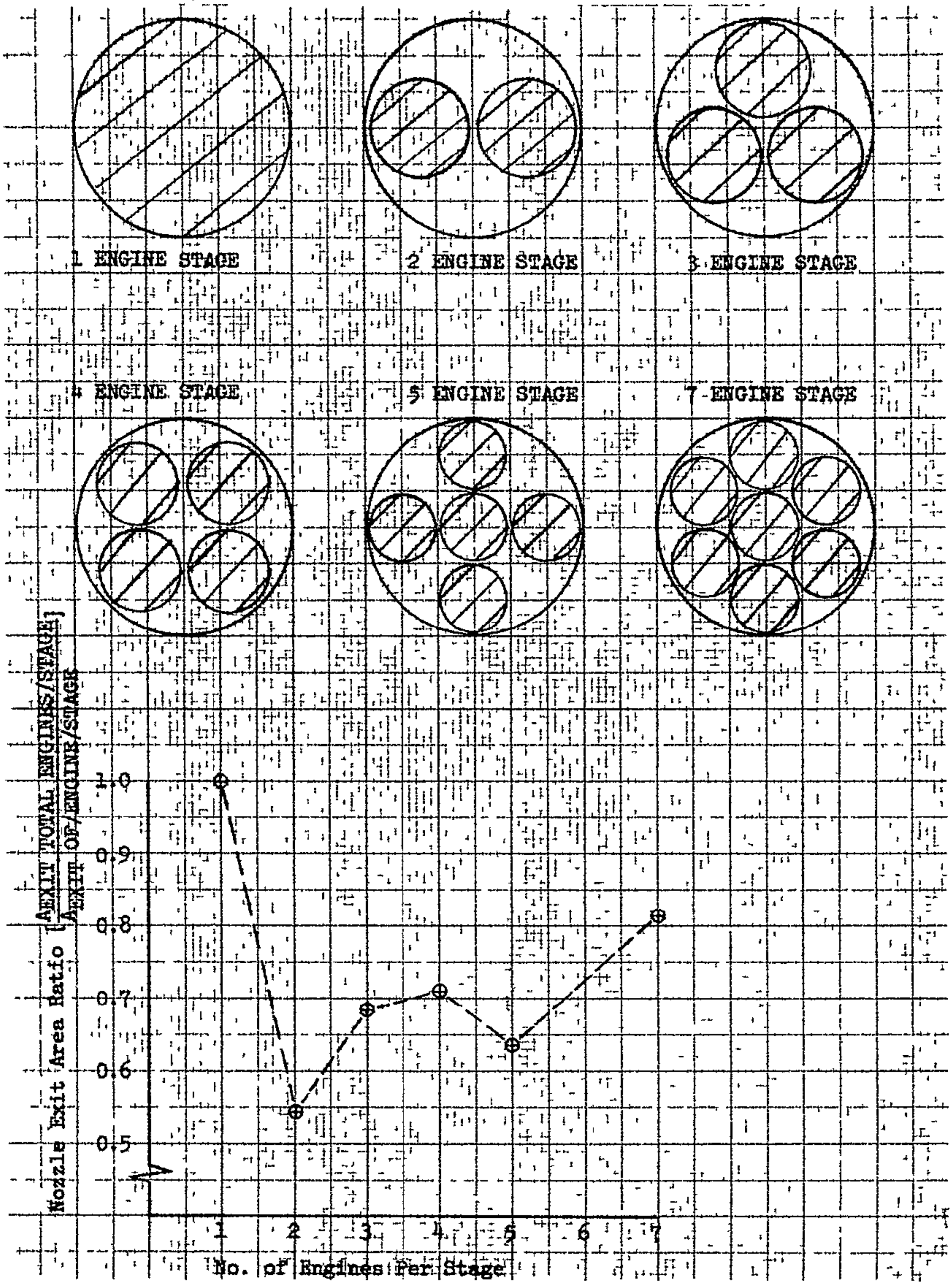


Figure 3-1 Clustering Effects on Nozzle Expansion Ratio

the propellant feed system and thereby reduces the recurring costs and, in the case of a pressure-fed engine system, permits some reduction in overall engine length and tankage weight by allowing the thrust chamber to be partially submerged in the aft propellant tank

Multiengine configurations, however, result in smaller individual engines thereby reducing engine scale-up development problems and reduce stage length with attendant interstage weight savings This provides a shorter vehicle length which would reduce the launch facility nonrecurring costs, i e , gantrey tower, assembly building, etc , however the recurring costs would be increased, i e , checkout, etc

After analyzing an initial vehicle candidate the first-stage weight was significantly higher than expected Upon investigation, it was found that the major contributor to this high weight was the engine Engine weights for the parametric analysis are derived from an emperical relationship (see equation 4-1, Section IV) This expression relates engine weight to basic design parameters such as thrust, chamber pressure, area ratio etc Since the exponent of thrust in this relationship is greater than one--single engine configurations, with their attendant higher thrust/engine, tend to yield higher engine system weights Weight data obtained from engine manufacturers tends to substantiate the judgment that, from an engine standpoint alone, a single engine weighs more than a cluster of engines for a given total thrust level An analysis was performed to determine the stage weight sensitivity to the number of engines The results of the analysis are shown in Figure 3-2, which depict normalized stage burnout weight (ratio of the burnout weight of a multiengine configuration over its single engine equivalent) as a function of the number of engines and propellant load It should be noted, that the dashed portions of these curves represent engine configurations which would exceed the diameter restraint that is nominally associated with the propellant load shown Even considering the geometry restraints, Figure 3-2 indicates a definite weight advantage for multiengine phase of the study, single pump-fed engine configurations would only be considered up to a thrust of 1 5 x 10^6 lb (A thrust of 1 5 x 10^6 lb is representative of the maximum existing pump-fed hardware) Single pressure-fed engine configurations were considered up to a thrust of 3 0 x 10^6 lb, which is representative of the low-cost pressure-fed

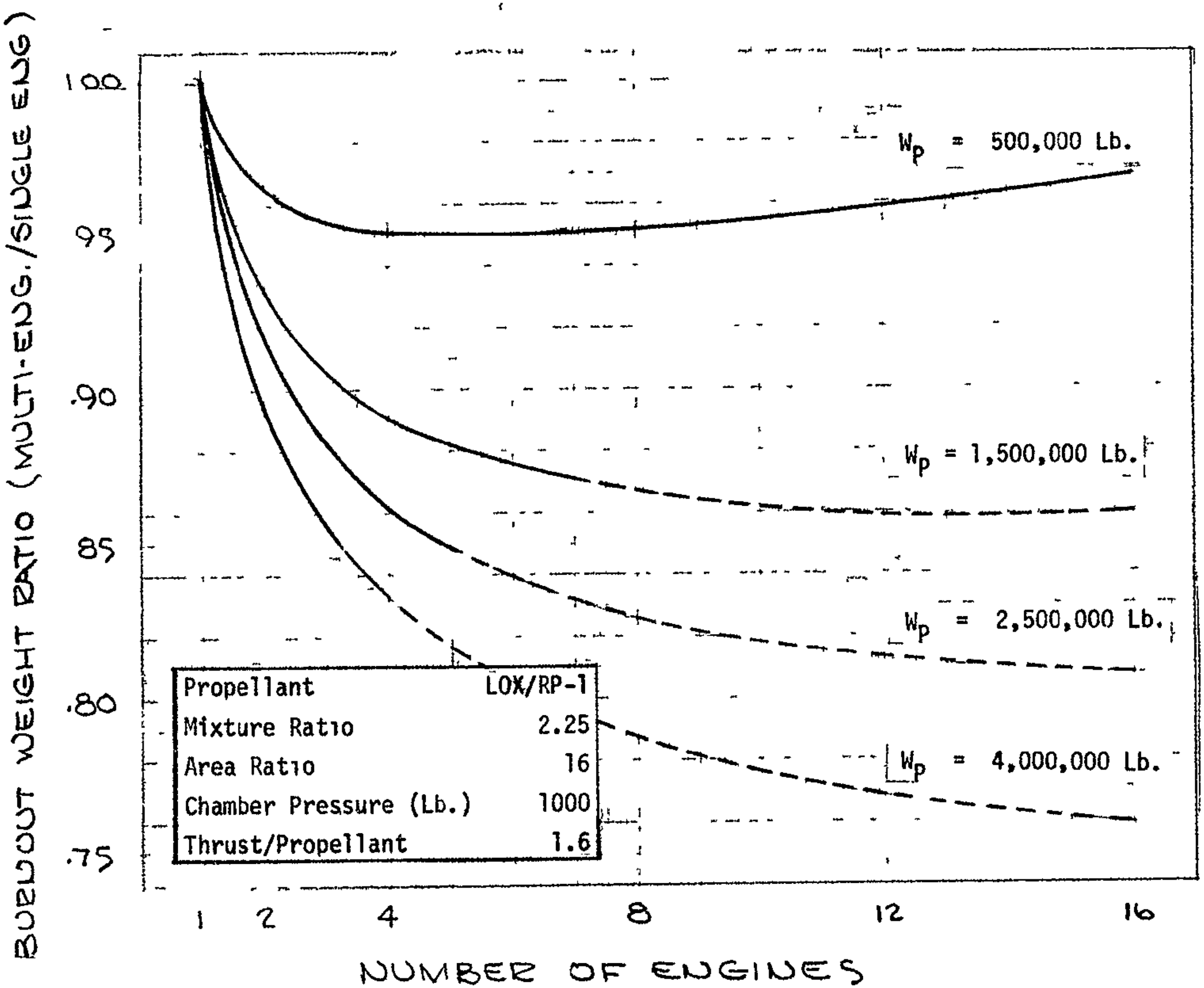


Figure 3-2 Weight Effect Multiple Engines (Pump-Fed)

engines currently being considered for development by AFRPL Stages requiring thrusts greater than these values utilized multiengine configurations with the individual engine thrust value being equal to or less than the appropriate system maximum thrust indicated above

3.1.1.4 Pressurization Systems - Shown in Figure 3-3 are simplified schematics of the pressurization system concepts selected for the pump-fed liquid propulsion candidates The salient features of these pressurization concepts are briefly discussed below

The pressurization concept selected for LO_2/RP-1 pump-fed candidates, utilizes gaseous oxygen to pressurize the oxidizer tank and ambient helium for fuel tank pressurization The pressurization system for the LO_2 tank uses gaseous oxygen to maintain the LO_2 tank pressure during the boost phase Gaseous oxygen is obtained from engine mounted heat exchangers which vaporize and heat high-pressure liquid oxygen from the turbopump This concept is identical to that used for the Saturn S-IB, S-IC, and S-II stages Pressurant bled from the engine system is an inherently simple concept which lends itself to the low-cost philosophy

Based on an LO_2 tank pressure of 50 psia and effective pressurant temperature of 385°R, the weight of gaseous oxygen in the ullage at stage burnout is

$$W_{GO_2} = 0.00545\, W_{LO_2}$$

where W_{LO_2} is the weight of oxygen expelled The pressurization system hardware is assumed to be negligible in comparison to the vapor weight for this type of system

A low-cost approach for pressurizing the RP-1 tank is not as straightforward as for the LO_2 tank A survey of existing RP-1 pressurization systems has revealed that an ambient gas blowdown system is about the most simple of current practices Therefore, this system was selected for the parametric analysis Based on a tank pressure of 50 psia and using helium as the pressurant, the gas weight is

$$W_{He} = 0.001306\, W_{RP-1}$$

where W_{RP-1} is the weight of fuel expelled Assuming high-strength steel bottles and an initial helium storage pressure of 3,000 psia, the weight of the helium storage bottles is

$$W_{Bottle} = 0.014\, W_{RP-1}$$

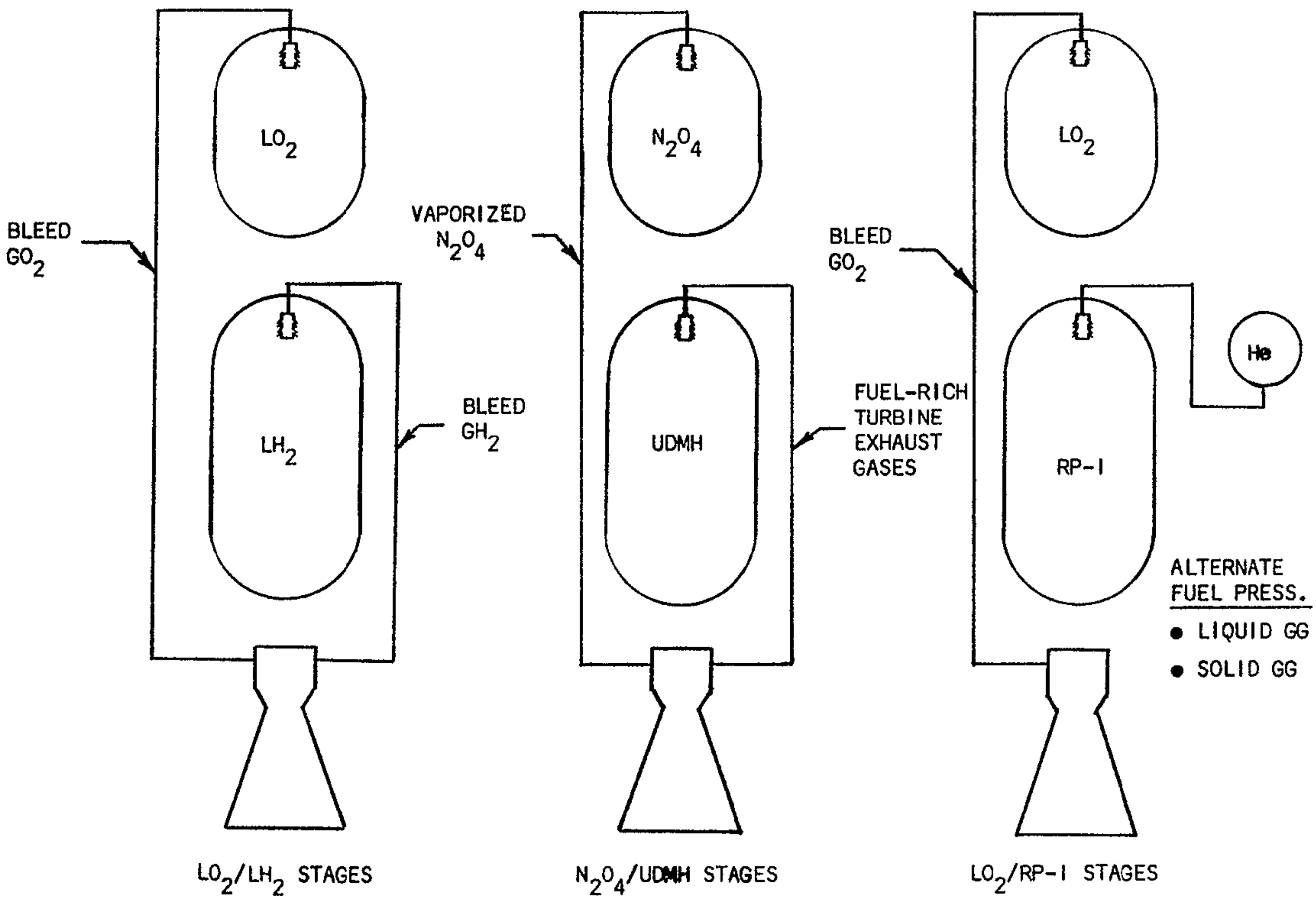


Figure 3-3 Pump-Fed Pressurization Systems

The parametric RP-1 pressurization system weight is the sum of the gas and bottle weights. Other hardware weight for the RP-1 pressurization system was assumed to be negligible.

The pressurization concept selected for pump-fed storable candidates, is the same as is used in the Titan launch vehicle. Oxidizer (N_2O_4) is bled from downstream of the oxidizer pump and vaporized by a heat exchanger located in the turbine exhaust. The vaporized N_2O_4 enters the oxidizer tank at 220°F. The fuel tank is pressurized by turbine exhaust gas which is cooled to 120°F in an exchanger located in the fuel supply line. This type of engine bleed-gas pressurization system is simple and lends itself to both high-performance and low-cost design philosophy. The pressurization system adopted assumes a tank pressure of 50 psia exclusive of fluid head. The tank residual vapor for a 50 psia tank pressure is

$$W_r = 0.0032\ W_p$$

where

W_r = residual vapor weight

W_p = total propellant aboard

The pressurization system selected for the pump-fed LO_2/LH_2 stages uses vaporized propellants as the pressurant. This approach also lends itself to a low-cost system. The propellants are vaporized by the engine gas generator exhaust products. Both M-1 and J-2 engines have this type of heat exchanger capability. The resulting vapor residuals, based on 50 psia and an effective pressurant temperature of 385°R are

$$W_{GO_2} = 0.00545\ W_{LO_2}$$

$$W_{GH_2} = 0.00574\ W_{LH_2}$$

Figure 3-4 shows the simplified schematics for the pressurization concepts which were selected for the pressure-fed liquid candidate stages. Following is a brief discussion of the selected systems.

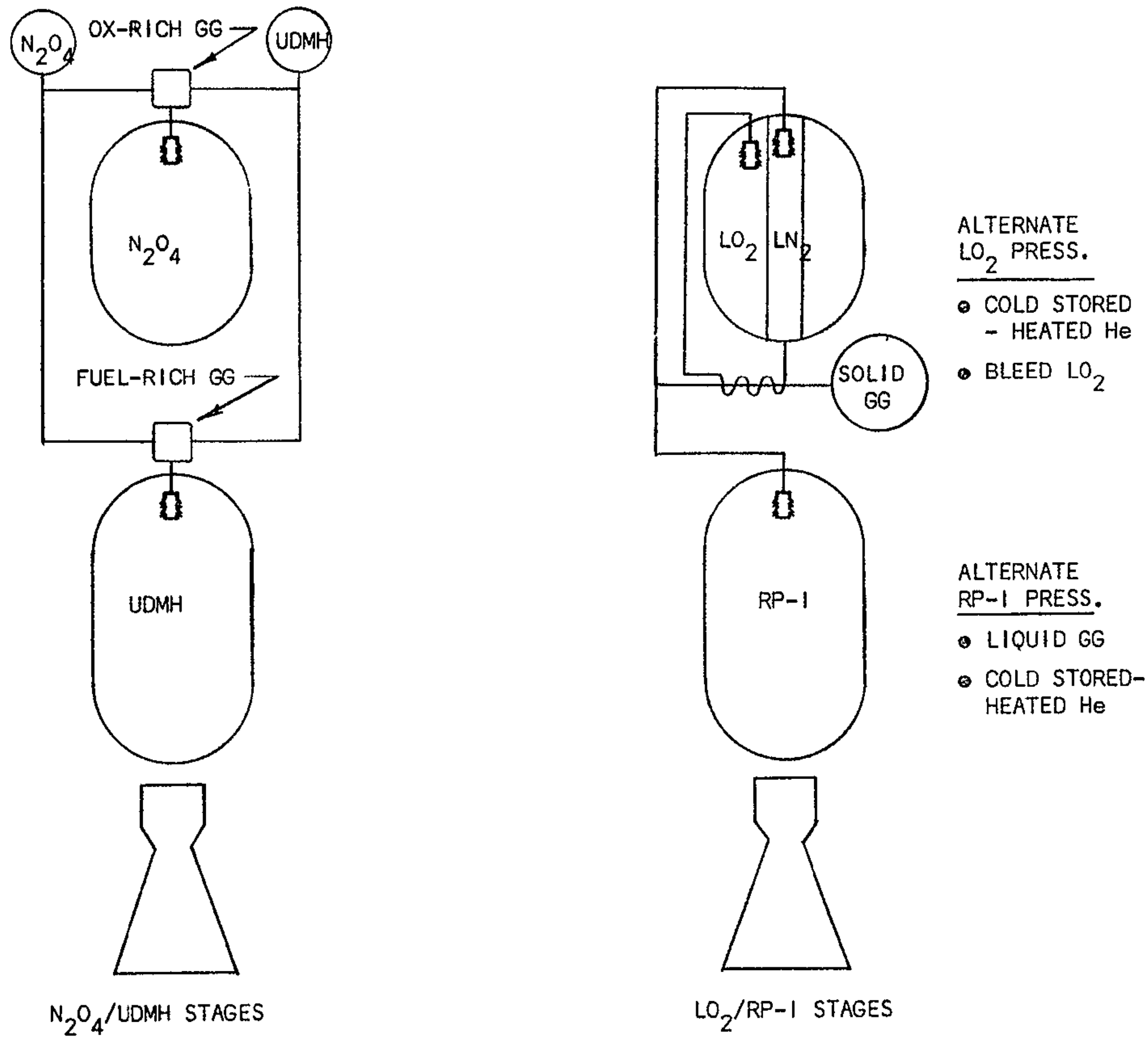


Figure 3-4 Pressure-Fed Pressurization Systems

For N_2O_4/UDMH pressure-fed candidates the tanks are assumed pressurized by individual bipropellant liquid gas generators Ambient-stored nitrogen pressurizes the auxiliary tanks The weights of the system are given as follows

	First and Second Stages	Third Stage
Weight of auxiliary propellant and tankage	$0\ 0121W_f + 0\ 0155W_{ox}$	$0\ 0081W_f + 0\ 007W_{ox}$
Weight of GN_2 and tankage	$0\ 00136W_f + 0\ 00129W_{ox}$	$0\ 00106W_f + 0\ 001W_{ox}$
Aux propellant tank pressure, psia (ref)	450	300

where

W_f = stage fuel weight

W_{ox} = stage oxidizer weight

This type of system is currently being considered throughout industry for application to low-cost pressure-fed storable stages Another approach which was considered prior to selection of the gas-generator concept was main tank injection (MTI) In this concept, fuel is injected into the oxidizer tank where hypergolic reaction takes place This reaction in the tank raises the tank pressure By proper control of the injection fluid, the tank pressure can be maintained at the correct level The fuel tank would be pressurized by injecting oxidizer in the fuel tank in a similar manner While MTI provides a light-weight and low-cost pressurization system, the feasibility of this concept for large stages is questionable pending the results of current development efforts Therefore, the bipropellant gas generator was selected because of the greater certainty associated with this concept

The selection of a practical low-cost pressurization concept for the pressure-fed LO_2/RP-1 stages was not as straightforward as for the other candidate liquid stages A cold-stored heated helium system was considered for pressurizing the LO_2 in a manner similar to that used on the Saturn S-IC and S-IVB stages Cold stored helium systems however are quite large for pressure-fed application due to the high tank pressures Consequently, the helium storage bottles become costly simply because of the large volumes which are required Therefore, a vaporized

LN_2 pressurization system was selected for the LO_2 tank In this approach, the pressurant is stored as a liquid which, because of the high liquid density, requires lower storage volume than for gaseous storage The LN_2 is vaporized and heated to about 400^oR by a heat exchanger mounted in the gas generator duct

The RP-1 tank is pressurized by the efflux of a solid-propellant gas generator Because the gas generator efflux is used to vaporize and heat the LN_2 for the LO_2 tank pressurization, the temperature of the GG efflux is reduced to about $1,200^oR$ before entering the RP-1 tank A portion of the GG exhaust gases is also used to expel the LN_2

The following pressurization system weight estimating relationships were used for the pressure-fed LO_2/RP-1 stages

	First and Second Stages	Third Stage
Weight of gas generator	$0\ 00258W_{LO_2} + 0\ 033W_{RP-1}$	$0\ 00125W_{LO_2} + 0\ 0222W_{RP-1}$
Weight of LO_2 system	$0\ 0447W_{LO_2}$	$0\ 0298W_{LO_2}$

3 1 1 5 Thrust Vector and Roll Control - A gimballed engine thrust vector control (TVC) system was selected for the pump-fed configurations and a liquid injection (LI) TVC for all pressure-fed configurations except those using LO_2/RP-1 propellants A gimballed engine system was chosen for the LO_2/RP-1 engine system Following is a discussion of the rationale for these selections

The pump-fed operating chamber pressure is significantly higher than its equivalent pressure-fed counterpart, thereby requiring a smaller chamber volume, hence providing a smaller mass to be gimballed Based on existing design data, this type of TVC provides the most cost effective system for the pump-fed configuration, and further, the required component designs are well within the current technology

An orthogonal LITVC system was selected for the pressure-fed engine systems (except those using LO_2/RP-1 propellants) as the most cost effective TVC system because of the on-board availability of the injectant fluid (N_2O_4) This approach was also selected for use on the low-cost pressure-fed engines currently being developed by TRW and Rocketdyne for AFRPL because of its minimum system recurring cost advantage

A gimballed engine system was selected for the pressure-fed engine stages which use LO_2/RP-1 propellant This stage does not have the advantage of on-board injectant availability, and, the use of a LITVC system would require the addition of the injectant, necessary tankage and pressurization system, etc which would result in increased costs

Analyses of the roll control systems (RCS) were not performed during the initial screening phase It was assumed that bipropellant or monopropellant RCS would be used on all single engine stage configurations and differential TVC system actuation for multiengine configurations to provide the necessary roll requirements It was assumed that the differential cost of these system concepts would have a minimum effect upon the total vehicle cost Therefore, the omission of the RCS costs should have little or no impact upon the relative study results

3 1 1 6 Propellant Management - Effective propellant management results in increased stage performance by reducing the propellant residuals in the tank at burnout Two basic approaches toward providing increased propellant utilization (PU) can use either open-loop or closed-loop systems In the open-loop system approach, high propellant utilization is achieved by predicting the in-flight propellant usage (taking into account deviation from the normal operating points) and accurately loading the propellants to achieve minimum residuals The closed-loop system employs on-board sensing devices which monitor the propellant remaining in the fuel and oxidizer tanks A control system is provided which adjusts the respective flow rate of the propellants to the engine to achieve minimum residual propellant The open-loop approach is desirable because it really involves no system at all, other than the propellant loading sensors However, its adequacy depends on how well the system variations and their influence on performance can be predicted The closed-loop approach usually results in very low residual weights at the expense of adding a rather complex and costly system to the vehicle

Current vehicles systems operating on an open-loop basis are obtaining PU on the order of 99-1/2% Closed-loop operation on the other hand has resulted in PU better than 99-3/4% Because of this slight difference and the significant cost associated with a closed-loop system, the open-loop approach has been used in this study To accommodate a low-cost philosophy in stage design, an open-loop PU of 99% was used to account for increased tolerances in the engine and feed system operation

3.1.2 Solid Propulsion Systems - Single motor stage configurations of 260-in diameter solid-rocket motors (SRM) were evaluated during the study using current technology Clusters of 156 in diameter solids were also considered, however, this perturbation was judged more appropriate to the detailed analysis portion of the study (note this portion of the study has since been cancelled) For the selection analysis, it was assumed that a single motor 260 in diameter solid was representative of solid motor concepts in general For example, the selected propellant, a rubber base composite (PBAN), was used in both 156- and 260-in SRM demonstration test firings and is currently being used in both the Titan III-C and Minuteman programs

The motor chamber pressure was selected from wind load structural considerations The maximum expected operating pressure (MEOP) was selected as the minimum value required to obtain the unpressurized case structural thickness capable of carrying ground wind loads The required MEOP determined in the subsequent analysis was 672 psia. This loading condition is discussed further in Section 3 3. The average pressure of 545 psia used to determine motor performance propellant loading, nozzle throat size, etc was obtained using the 672-psia MEOP value and typical pressure design factors

Although no attempt was made to optimize the system parameters nor to perform subsystem tradeoff studies, the system selections were based on previous experience derived from related hardware programs and studies The motor operating parameters used in the SRM weight and sizing analysis are shown in Table 3-3 A typical motor configuration is shown in Figure 3-5 The resulting motor component launch weight summary for a propellant weight range of 0 5 x 10^6 to 8.0 x 10^6 lb is shown in Figure 3-6 The lowest propellant weight considered was 0.5 x 10^6 lb. This amount of propellant can be packaged in a 260 in spherical motor, using the assumed operating parametere, and therefore, a lesser amount of propellant would require appropriate off-loading It was therefore assumed that if a lesser amount of stage propellant was required, a more optimum design would be obtained using a smaller motorcase diameter

The motor lengths were determined from an assumed grain port to nozzle throat area ratio of 1.2, with an additional 5% off-loading in the aft head to facilitate nozzle assembly and to minimize gas-erosion problems. While no specific L/D limitation was imposed upon this analysis, the required case length for the large propellant weights, i e , 6 x 10^6 lbm, - L/D is approximately 9.5 It is not recommended that motor L/D's much greater than 9.0 (7-segment, 120-in diameter motor) be used until more detailed vehicle configuration analysis is performed

Table 3-3

SOLID ROCKET MOTOR OPERATING PARAMETERS

Stage	First	Second	Third
• Chamber pressure, psia			
Average	545	545	545
Maximum	600	600	600
MEOP	672	672	672
• Area ratio (conical nozzle)	5	10	10
Nozzle divergence angle, deg	15	15	15
• Specific impulse (S L) sec	231	---	---
• Specific impulse (vac) sec	252	268	268
• I_s efficiency	93%	93%	93%
• Burn time, average, sec	153	153	153
• Port-to-throat ratio	1 2	1 2	1 2
• Case material	← 250 grade maraging steel →		

3 1 2 1 Propellant - The propellant considered for this application was a current technology polybutadiene - acrylonitrile - acrylic acid/ammonium perchlorate/aluminum (PBAN) type whose characteristics are listed in Table 3-4 Since these characteristics are generally typical of this type of propellant, it was assumed that the required burn rate was available or could be obtained with a minimum of formulation effort. The maximum web fraction (ratio of effective web to grain outside radius) obtained was 0 85 for the 0 5 x 10^6 lbm propellant weight motor which is considered to be within allowable limits for this application

3.1.2.2 Case - The case thickness was sized for internal pressure loads using the maximum expected operating pressure (MEOP), a structural safety factor of 1.25, with 250 grade maraging steel (ultimate stress = 250,000 psi). The cased heads are hemispherical which are welded to a continuous momocoque barrel section The case weight curve shown in Figure 3-6 reflects the above parameters, however, it does not include skirts nor case integral skirt rings

The case insulation size and associated weight was determined using 0 008 and 0.018-in. erosion rate in the forward and aft head, respectively, in addition to 0.2-in. insulative thickness. No attempt was made to optimize the insulation design.

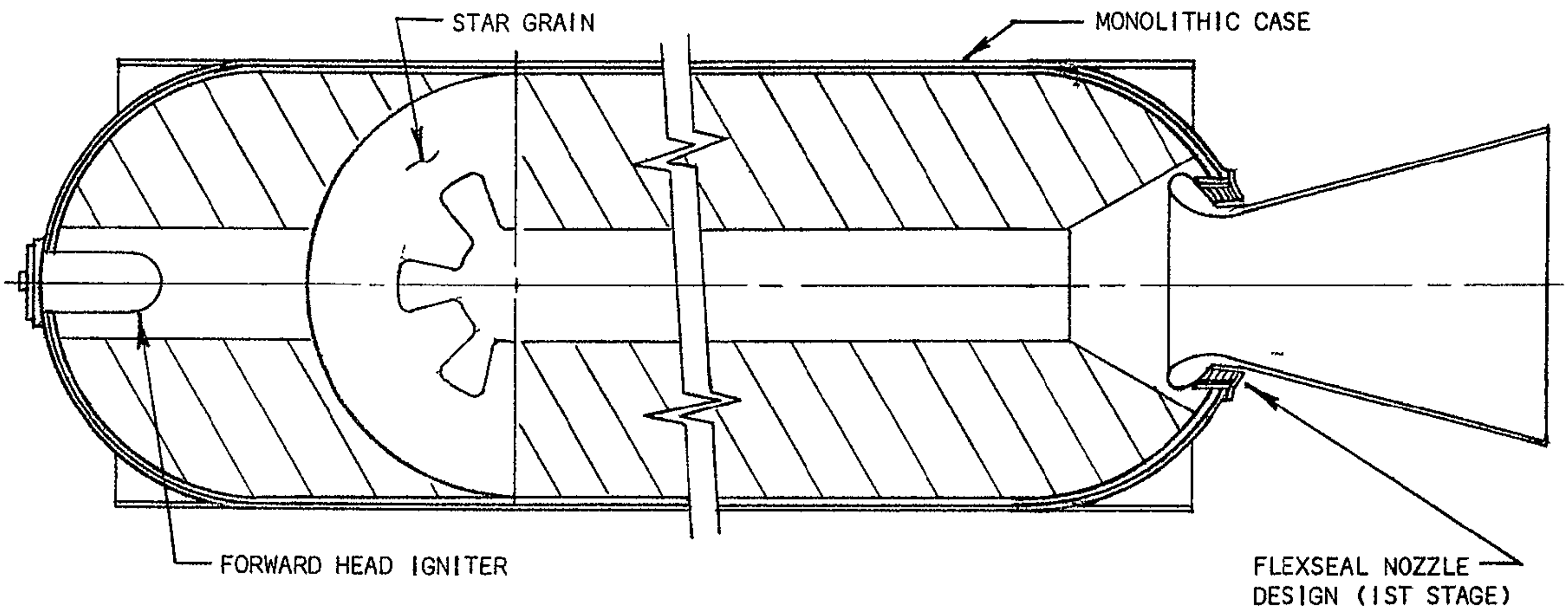


Figure 3-5 Typical 260" Diameter SRM Configuration

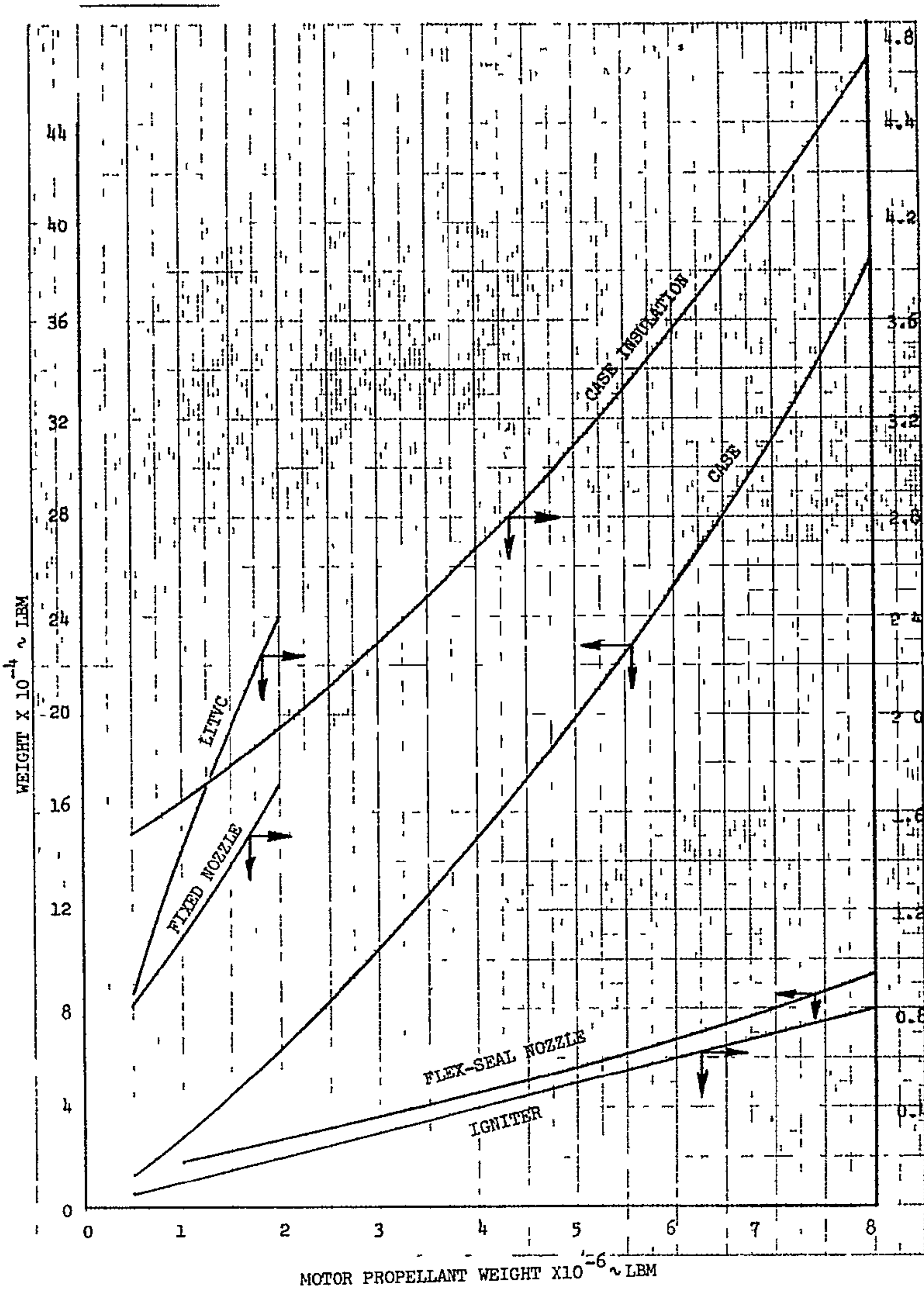


Figure 3-6 260 SRM Launch Weight Summary

Table 3-4
PROPELLANT CHARACTERISTICS

Type	Composite (Class 2)
Characteristic Exhaust Velocity, fps	4,980
Ratio of Specific Heats	1 18
Density, lb/in ^{3}	0 064
Combustion Efficiency, %	95
Specific Impulse, std (1,000, 14 7, 0°), sec	248

3 1 2 3 Nozzle - The nozzle design was a conventional ablative type using a steel shell and ablative throat and liner material The nozzle expansion ratio (ϵ) was selected, based on previous studies, as 5 for the first-stage and 10 for the second- and third-stage motors In order to eliminate costly aft first stage motor flare structures and to facilitate stage-to-stage assembly, the nozzle exit diameter was to be maintained smaller than the motorcase outside diameter The motor propellant weight limitation resulting from this constraint is 5 25 x 10^6 lbm for first- and upper-stage motors respectively can be obtained with for the selected nozzle area ratio The corresponding case L/D for the 5 25 x 10^6 lbm weight is approximately 8 0 and is consistent with current solid-motor practices

3 1 2 4 TVC - Based on the results of previous MDAC studies, a "flex-seal" movable nozzle was selected for the first-stage motor and a simplified orthogonal liquid injection LITVC system for the second- and third-stage motors as being the most cost-effective systems The side force duty cycles which were used in sizing the TVC systems are shown in Figure 3-7 These include 0 5° jet deflection to compensate for thrust misalignment and CG offset in addition to wind loads (first stage only) and stage separation induced loads

The TVC system weight curves shown in Figure 3-6 include the auxiliary power unit (APU) weight as well as two linear actuators for the movable nozzle system The APU considered for movable nozzle system uses a continuous burning warm solid gas generator to pressurize the hydraulic fluid used to actuate the nozzle A typical movable nozzle system design is shown in Figure 3-8 The APU considered for the LITVC system uses an electric motor-driven pump to

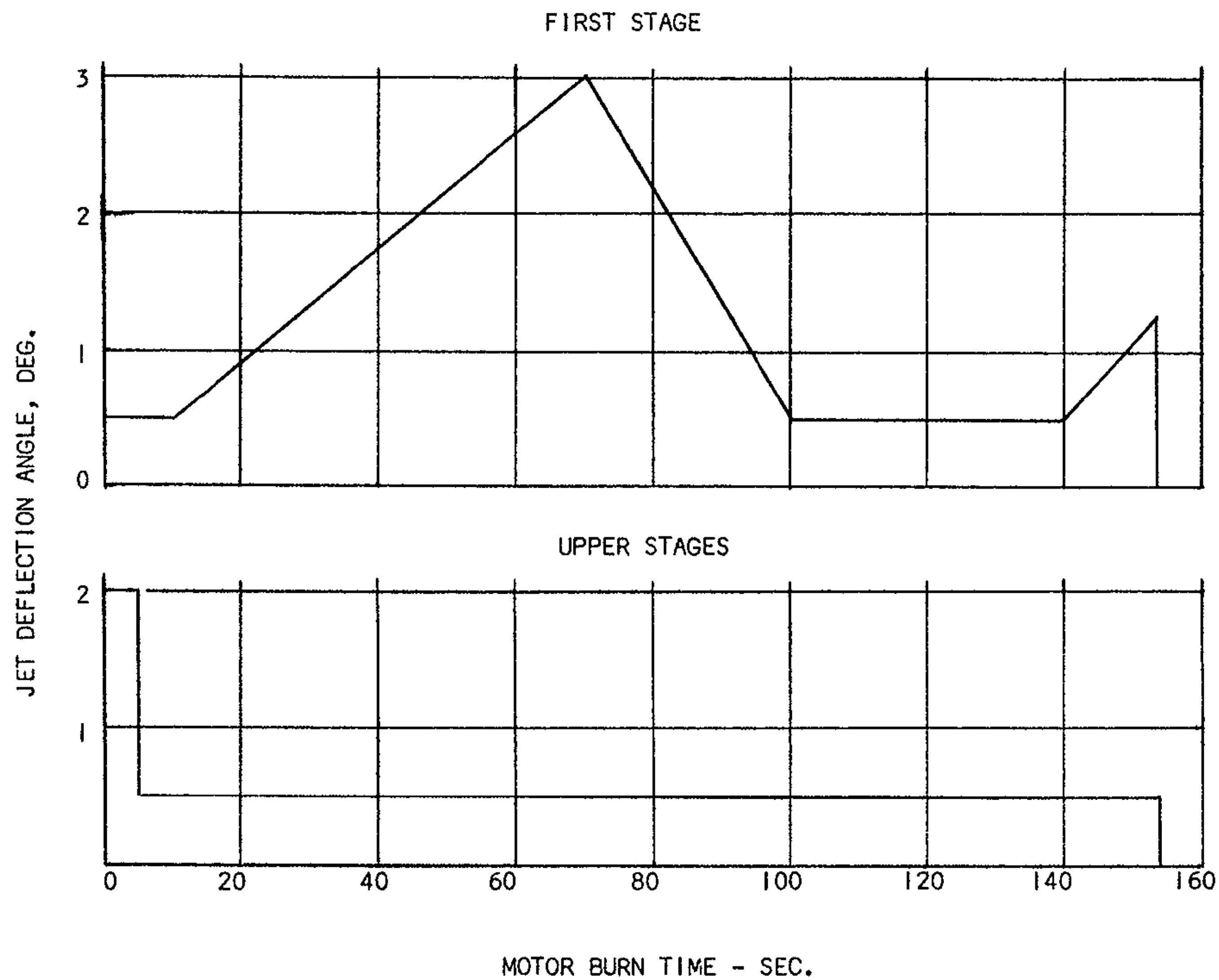


Figure 3-7 TVC Duty Cycle

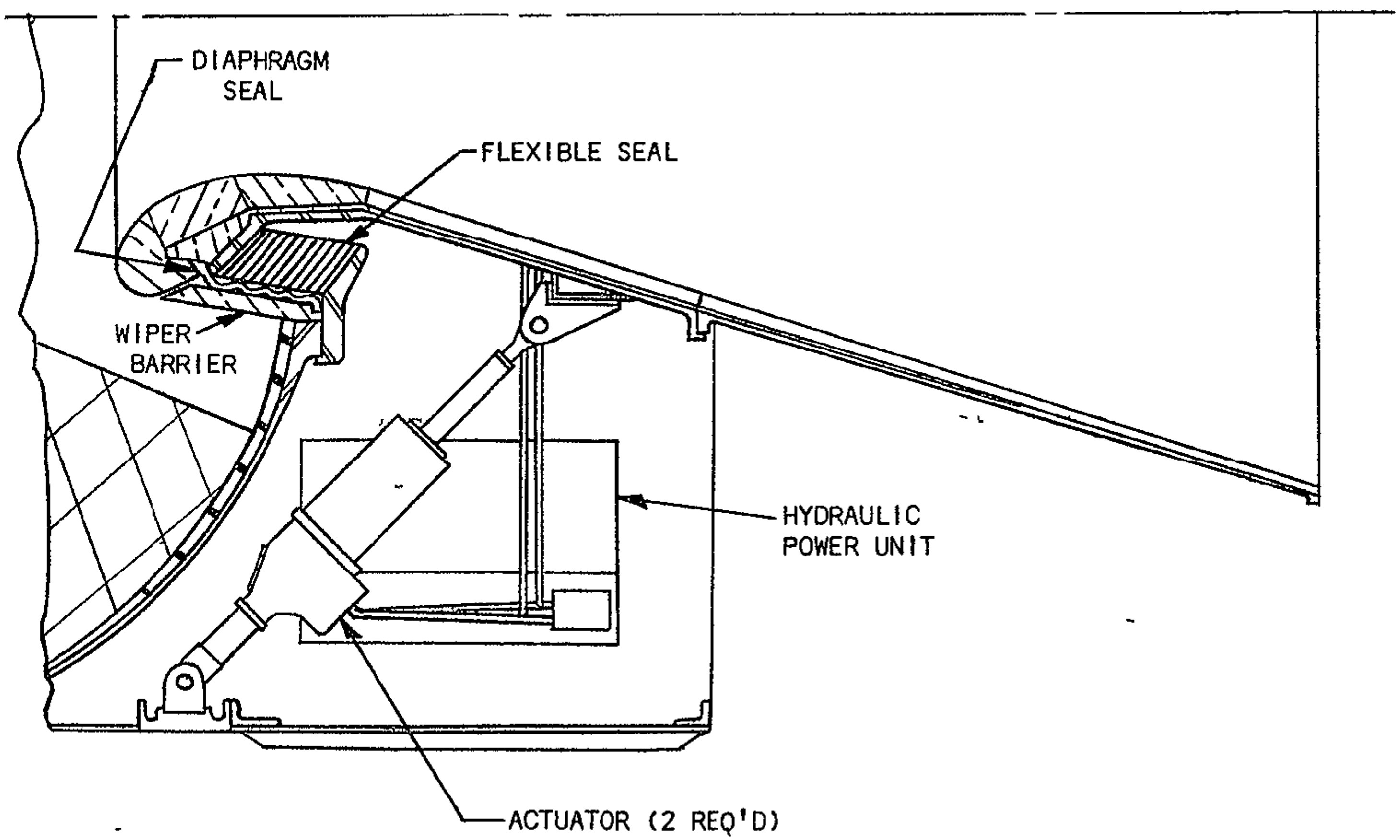


Figure 3-8 Movable Nozzle Design

pressurize the hydraulic fluid used to actuate the injectant valve poppet This system is very similar to the operational APU used on the Titan III-C vehicle A LITVC system schematic is shown in Figure 3-9

3 1 2 5 Expended Inert Weight - An estimate of inert motor component weights expended during the firing duration was made to permit a more realistic evaluation of vehicle performance at stage burnout This allows degradation of the motor specific impulse resulting from the addition of the higher molecular weight, lower heat capacity inerts into the gas exhaust stream while realizing lower vehicle burnout weight values Inert weight losses, as a percent of component weight at launch, are as follows

Case insulation	41% of component
Igniter	42% of component
Fixed nozzle (ϵ = 10)	1 5% of component
Flex-seal nozzle	15% of component
LITVC	65% of component

With the exception of the igniter, these weight losses were considered linear with respect to motor burn time The igniter weight loss was assumed to occur within 5 sec after fire-switch

3 2 Electromechanical Systems Definition - The principal objective of the electromechanics subsystems task in this study was two-fold (1) evaluate criteria and requirements and determine an overall system concept reflecting the optimum relationship between cost effectivity and functional reliability and (2) size and define specific systems for the candidate vehicles to the extent required to aid in candidate selection by inputing weight parameters to performance analyses and cost factors to pricing evaluations

The collection of vehicle subsystems identified as avionics, astrionics, flight electronics, etc , are primarily electromechanical communication and interface devices They are generally not requirements themselves but rather solutions to the functional requirements of the remainder of the vehicle They provide the means of translating mission instructions into stimuli which control vehicle performance, detect deviations from instructions, generate commands which correct errors or sequence events, and keep man apprized of existing and impending conditions In performing these tasks the astrionics must interface with all other functional components which make up a total vehicle system

Experience has pointed out several significant facts pertinent to avionics systems

1 They occupy approximately 3% of the volume, contain less than 10% of the dry mass, and incur up to 50% of the total cost of currently existing launch vehicles

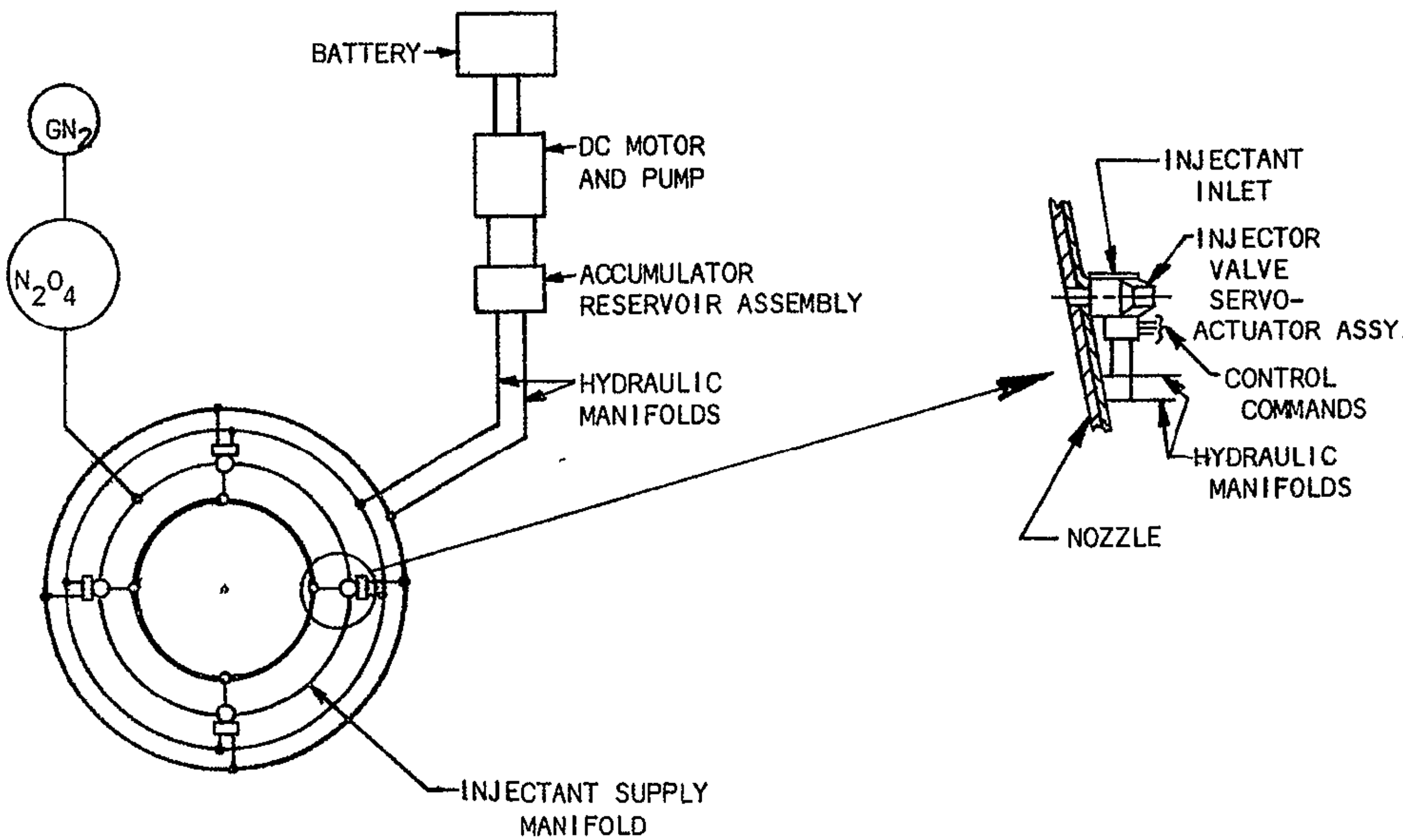


Figure 3-9   LITVC System Schematic

2. The ratio of labor cost to material cost is much higher in avionics than other fabricated systems such as structure, thereby reducing the significance of utilizing low cost raw material
3. A large portion of the true recurring costs of avionics is hidden under other categories, such as launch cost in the form of checkout, calibration, and data reduction
4 The size and complexity of the avionics system is not related to vehicle size but to the extent of the functional requirements generated by the vehicle designers.

If these facts are summarized into a criteria for low cost, it is immediately apparent that a different approach to cost reduction is required

Since the use of low-cost raw materials has little effect on total avionics system cost, one obvious solution is to reduce the size of the system (In fact, labor skill levels associated with difficulty of access and corrective rework resulting from repeated failure of marginal components could increase cost by an order of magnitude) System size can be reduced by two basic techniques (1) reduce the vehicle functional requirements which engender avionics subsystems and (2) employ subsystem components which can fulfill a number of similar requirements

The subsystems of an astrionics system are designed to fulfill requirements (real or imaginary) of the vehicle and the mission These requirements fall into the following basic categories and functions:

1 Guidance — Interfaces the vehicle to the mission profiles
2 Control — Sequences and quantitizes vehicle reactions
3 Instrumentation — Monitors vehicle conditions and performances
4 Telemetry — Interfaces vehicle with ground controls
5 Range Safety — Interfaces destruct ordnance to ground commands
6 Navigation — Interfaces vehicle guidance with ground trackings
7 Ordnance — Vehicle pyrotechnics for destruct, ignition, and stagings
8 Power — Supplies electrical energy to total vehicles
9 Distribution — Interfaces total vehicle electromechanical components

The first important step in creating an astrionics system which is both low cost and technically adequate is to ascertain the realistic functional operations which will be required of the system by the vehicle configuration and the proposed mission profiles These basic requirements must be reached by considering only an operational vehicle configuration. Each major-vehicle nonavionics system generates some minimum level of electromechanical interface which must be maintained if the system is to meet its mission objective For example, engines must be started and stopped, thrust vectoring systems must be actuated relative to trajectory, parameters which are accommodated by controlled system variables, or the effect of abort decisions must be monitored During the early phase of the study, these minimum requirements were determined for each candidate vehicle configuration and mission profile They were assembled in tabular format in groups which exhibit similar characteristics, such as "Requires In-Flight Computation," "Chronologically Real-Time Significant," "Stage Peculiar," "Mission Related," "Abort Decision Requirement," "Flight Evaluation Requirement," "Launch Operations Requirement" etc The first assembly iteration was coordinated with the other technical disciplines Vehicle and mission configurations were analyzed to verify that all requirements were considered and that no unnecessary functions were included The finalized tabulation provided the basis for sizing the capability of an astrionics system

In Aerospace systems, the primary consideration is function, however, secondary constraints of size, weight, and cost limit the design latitudes Astrionics devices have high density and are expensive The silhouette of a vehicle, which defines its volumetric capacity, is primarily restricted by propellant quantities With astrionics components the principal limiting factors are weight and cost since they can expand volumetrically without perturbating the vehicle size McDonnell Douglas Astronautics Company (MDAC) has made several Company analyses of current aerospace applications of the electromechanical disciplines oriented toward low-cost launch vehicles and has evolved criteria for obtaining lower system costs Five basic ground rules have been observed

- Development, qualification, and support costs, equal hardware costs for less than 30 systems
- Volume is inversely proportional to cost for a given approach to a functional system
- Interface between electronics systems is a very significant factor in system cost
- More capability than can be effectively utilized is the largest unnecessary cost in astrionics systems
- Checkout, calibration, and data management incur a large portion of astrionics system costs

Present aerospace program approaches utilize several early flights as R&D development tools thus necessitating one-for-one monitor and control functions for each significant variable on the stage This leads to immense instrumentation and calibration systems When the subject vehicle becomes operational, the high nonrecurring penalty (as stated 30 x recurring costs) for replacing harnesses and signal distribution systems with new designs tailored to the reduced requirements generally rules out any major modifications Figure 3-10 shows the relative cost rates between electro/electronic and nonelectrical systems for several representative launch vehicles and reflects the ratio that was considered desirable to achieve for low-cost launch vehicles

To make maximum utilization of the study effort in actual system definition and sizing, experience gained on the Thor Delta program was tentatively selected as representative of a good example of a starting point for system evolution Figure 3-11 gives a more detailed breakdown of astrionics cost factors between the totally operational, highly cost effective vehicle (MCD-Thor/Delta) and a typical space vehicle with a strong R&D background

Immediately apparent is the fact that in the strictly operational design, electro/electronics account for considerable less than half the vehicle total costs

3 2 1 Cost Effective System Definition - The electromechanical systems described in these data are for an operational vehicle which has been properly designed to allow removal of R&D subsystems after initial test flights, without perturbing the basic configuration The operational instrumentation and telemetry system will be limited to those measurements required to make mission decisions

The technology base used in the philosophical system layout for the vehicle is assumed to be available in the mid-1970's and emphasizes modular design, increased reliability, and low basic cost Interconnecting harness will, in all probability, be a flat-cable system with limited modification flexibility

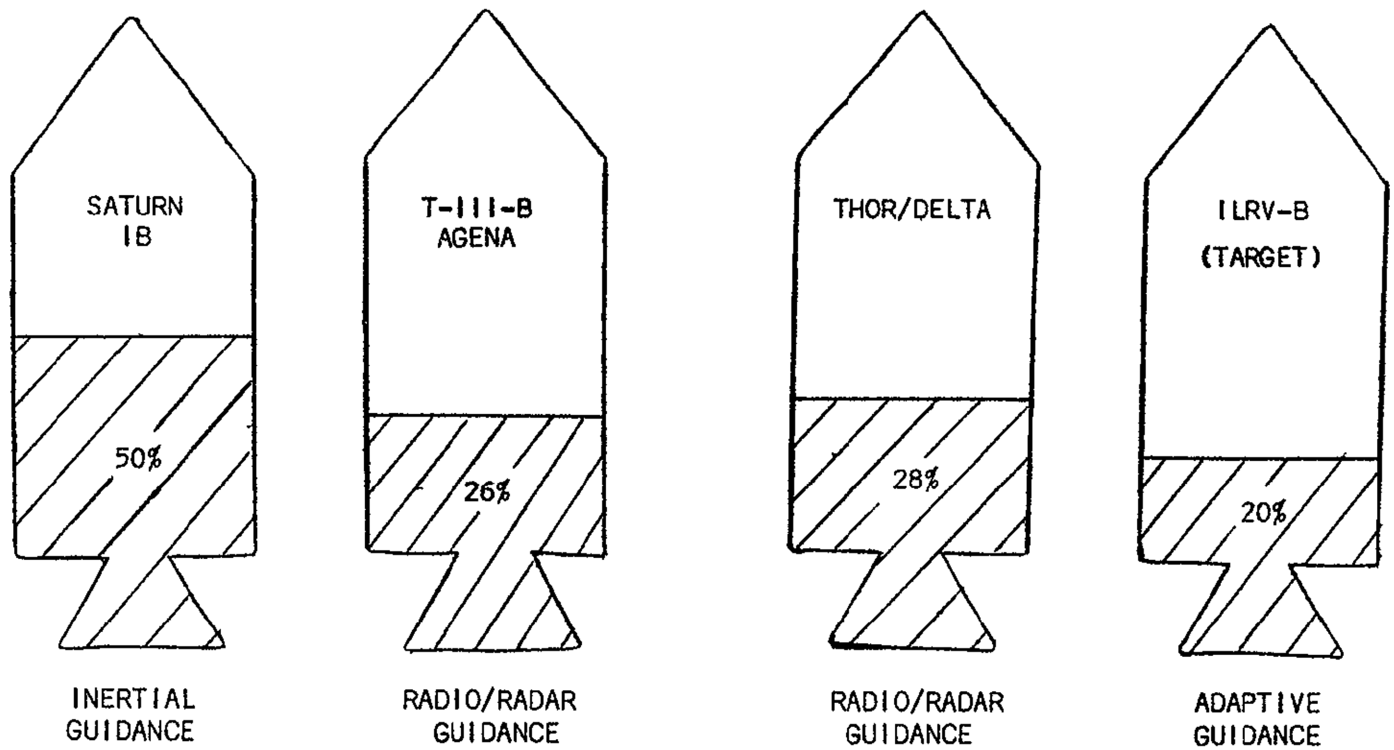


Figure 3-10 Astrionics Portion of Various Vehicle Costs

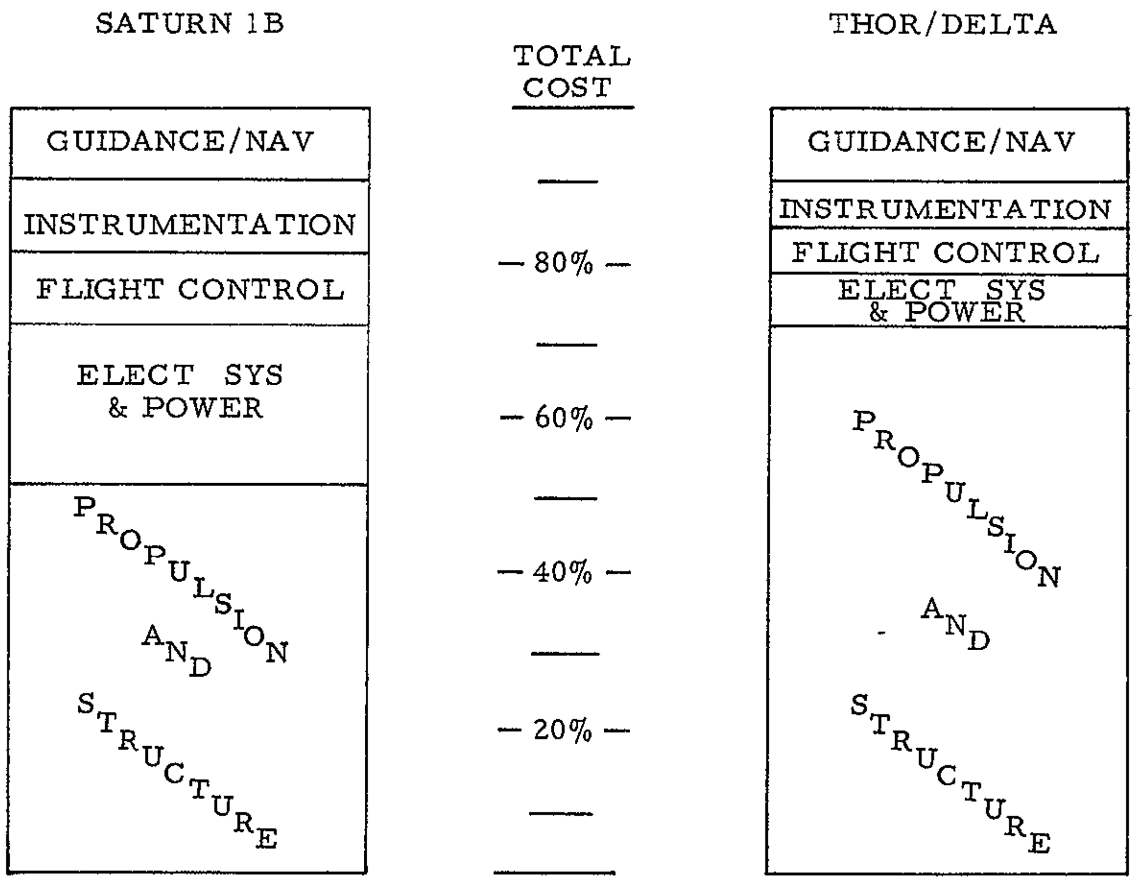


Figure 3-11 Major System Cost Ratio Comparison

The missions considered were low Earth orbit boost of a payload containing a master sequence clock and an active guidance system capable of producing the necessary precision in pitch, yaw, and roll error signals for the launch vehicle control systems to meet injection accuracy requirements A programmer/autopilot function is included in the final stage of the launch vehicle to provide interface between mission time base, guidance error signals, the vehicle operational sequence, and reactive force vectoring

The system philosophy described here is for an operational vehicle receiving guidance error and time synchronization signals from the payload It assumes a self-sufficient autonomous payload requiring only acceleration and attitude control forces from the launch vehicle during the boost/injection mission phase

3 2 1 1 Guidance and Control (G&C) Tradeoff - The first problem approached was to determine the best method of integrating guidance signals from the payload with control servo loops in the various boost stages Arguments can be presented pro and con for a modularized G&C system having certain functions retained in the lower boost stages as opposed to the total system centralized in one area In theory, a centralized system appears to be the optimum solution with all major astrionics components in the uppermost vehicle segment (i e payload) However, in practice this often breaks down as is evidenced in the Saturn family with the centralized IU In the case of ILRV, the large number of possible candidates exhibited many dissimilar control characteristics It was possible to have a three-stage vehicle with three different TVC systems entailing three types of steering equations and signal format to achieve adequate roll control and heading

Some serious cost problems arise from trying to combine vehicle functions with diverse formats into one centralized system The alternate solution, not to incorporate them, was a major area to be evaluated in this study Current technology and previous experience in space boosters indicate that one strong candidate solution to the control problem would be to have small modules in each stage Considering this possibility, the control problem was divided into several categories Criteria for division were such things as signal format uniqueness to a stage or repeated in several stages, requirements for guidance calculations, etc Centralized versus separated systems in terms of capability, installation, interface, checkout, and hardware costs were compared during the evaluation

As a result of the preliminary investigation, it was determined that by assuming a constant guidance and control format from the payload, and interpreting the commands as required in the upper stage of each vehicle configuration, a more equal comparison of stage requirements and performance could be obtained Figure 3-12 depicts the basic G&C components that will be located in the launch vehicle

3 2 1 2 Instrumentation System Description - A second major departure from past philosophy relates to the instrumentation and telemetry system The ratio between development costs and hardware costs of electromechanical systems (approximately 35 1) makes major modification after initial test flights fiscally unattractive Therefore, many R&D instrumentation systems interconnecting harness and signal conditioning modules remain in place on operational vehicles because it is impractical to redesign the hardware for a lower scale measurement program For the ILRV stages, a completely operational instrumentation and telemetry system will be installed from the first flight Any additional R&D measurements will be made with a totally autonomous instrumentation subsystem which can be added to early vehicles as a kit containing all necessary power sources, sensing, signal conditioning, encoding, and transmission These R&D kits are not considered in this weight and cost effort A FCC dictated shift to 2 5 gigahirtz TM frequencies in the 1970's will allow lower vehicle transmitter output power commensurate with the increased gain available at ground receiver antenna arrays This affords a reduction in electrical energy source requirements as well as reduced regenerative thermal output adding to the practicability of inflight passive environmental conditioning

For this study, the techniques of instrumentation and telemetry available in the 1970 time frame were analyzed to determine the advantages and disadvantages of each from a cost effectivity standpoint Multiplexing, encoding, modulation, and transmission systems were reviewed and the results tabulated in graphic format depicting cost per measurement versus quantity of data Figure 3-13 summarizes the results of the review Included in the basic cost are the heretofore generally neglected costs of reducing and compiling data after the flight Flight data reduction and management expenses are real recurring costs associated with the avionics systems of a launch vehicle In general these costs have been ignored but preliminary investigation has shown that on cumbersome R&D telemetry transmissions, the ground data management costs can exceed $5,000 per

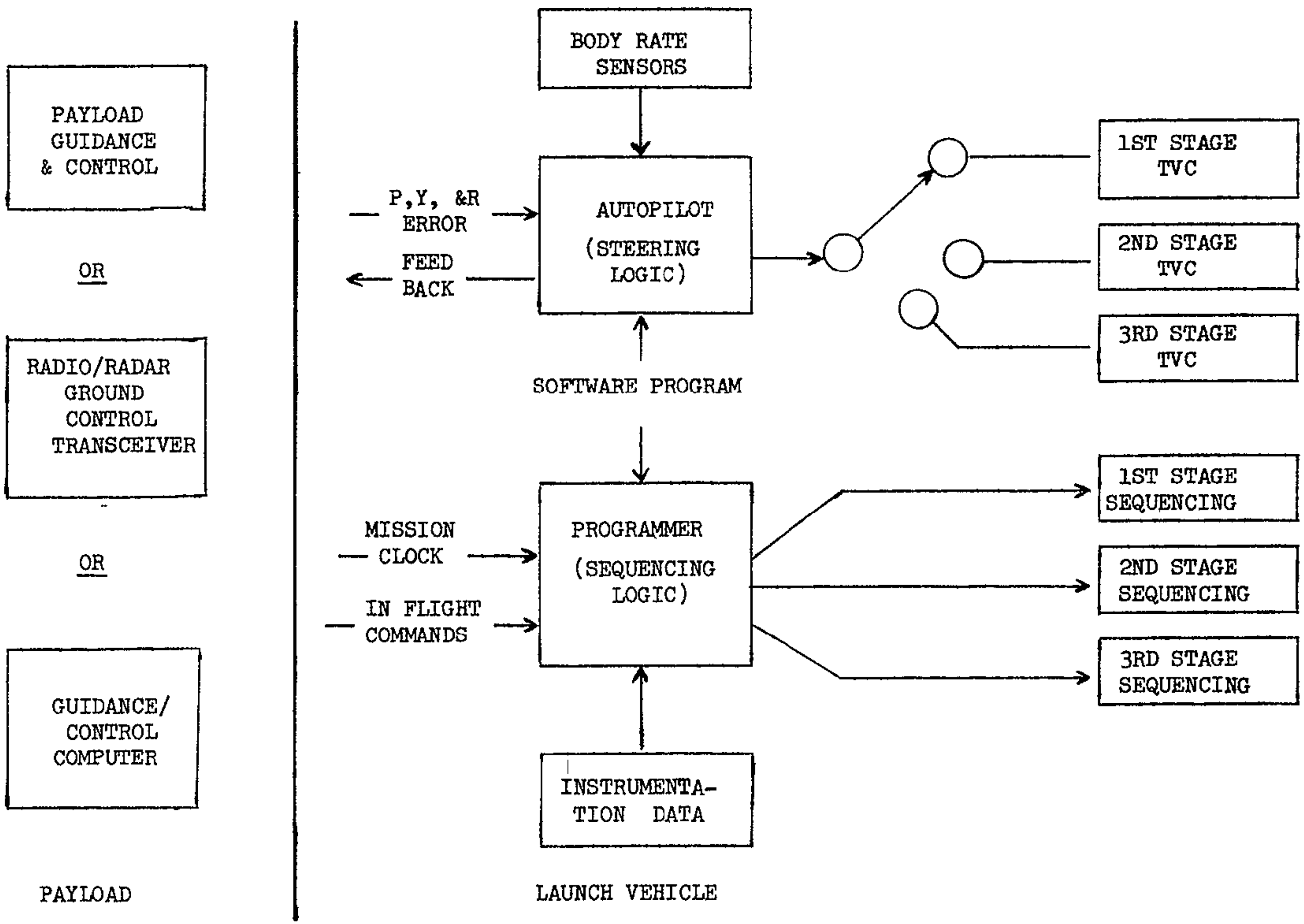


Figure 3-12 Basic Components of an Adaptive G&C System for ILRV

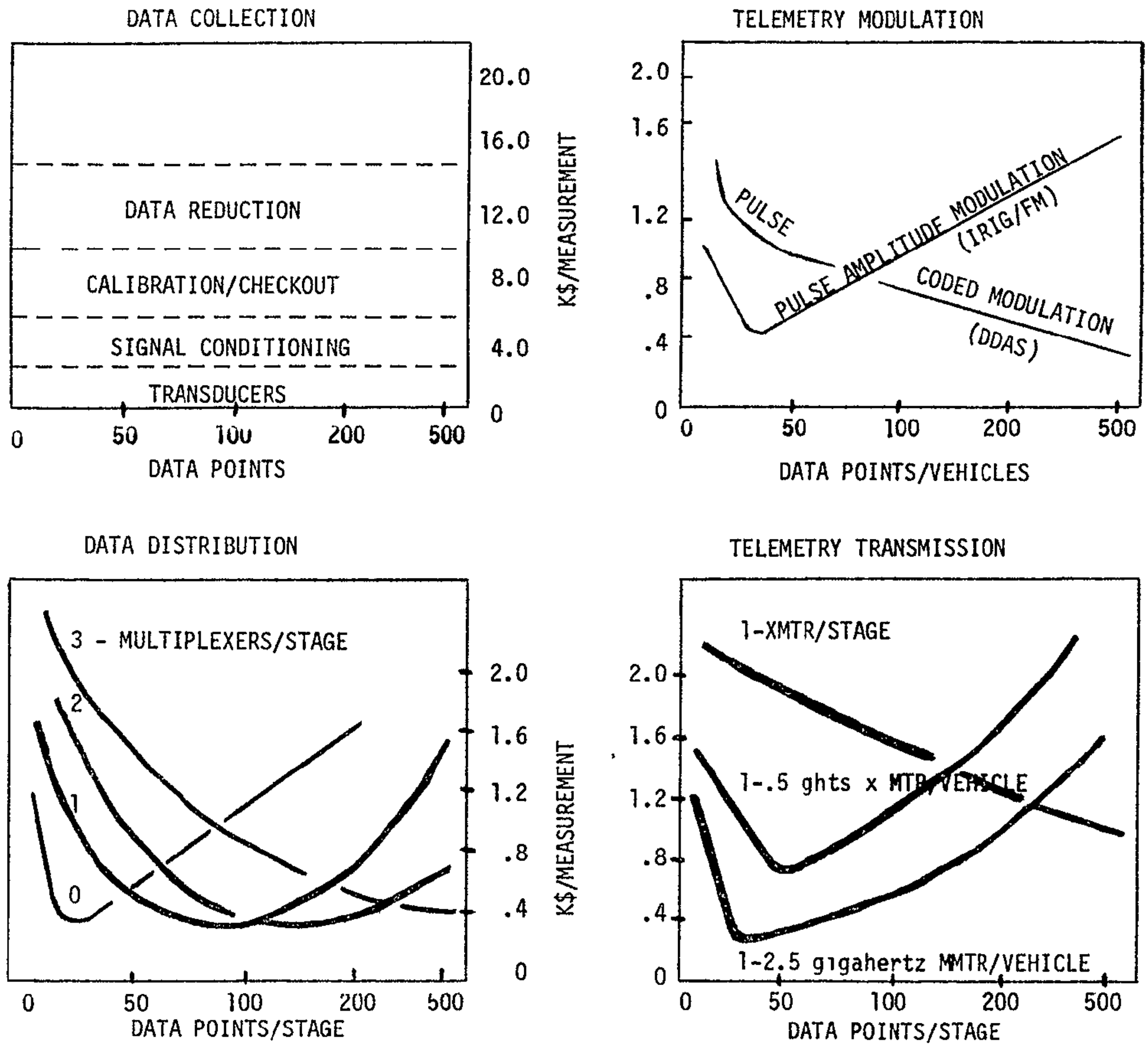


Figure 3-13 Cost Relationships of Instrumentation Techniques to Quantity of Data

parameter New and developmental instrumentation/telemetry concepts were investigated for applicability in the mission time frame under consideration Some basic areas for further consideration are

- The inclusion in avionics modules of a minimum of logic summing circuitry which will produce a single monitor signal representing module functional performance thereby decreasing the quantity of measurements required to verify proper operation
- The FCC dictated shift to 2 5 gigahirtz telemetry frequencies in the 1970's could allow lower vehicle transmitter output power commensurate with the increased gain available at ground receiver antenna arrays This affords a reduction in electrical energy source requirements as well as reduced regenerative thermal output adding to the practicability of inflight passive environmental conditioning

In order to make valid instrumentation system concept selections based on the results of the reviews of available techniques, a quantitized estimate of data points and types had to be formulated Each of the major disciplines was consulted and past vehicle experience was researched to arrive at a reasonable value for the number and type of measurements that could be expected on boost vehicle stage combinations Table 3-5 shows the results of this compilation and it can be seen that a two-stage vehicle should require approximately 100 data points while a three-stage vehicle needs about 150

Table 3-5

ILRV INSTRUMENTATION BREAKDOWN EXPENDABLE LAUNCH VEHICLES

Items	First Stage	Intermediate Stage	Terminal Stage	Totals
(*)Propulsion	24	24	10	58
Structures	2	2	4	8
Guidance & Control	6	10	26	42
Power	4	8	10	22
Emergency Detection and Range Safety	4	6	10	20
Stage Totals	40	50	60	150

(*)Determinations based on four engines each for first and intermediate stages One engine on terminal stage plus roll control motors, retrograde thrust on all stages

These numbers were then correlated to the results of the technique reviews (Figure 3-13) and the final system concept selection was made The instrumentation system consists of

- One 2 5-gigahertz PCM/FM transmitter in the final stage of the vehicle
- One PCM/DDAS modulation unit in the final stage
- One multiplexer per vehicle stage

Expected cost of the system is demonstrated in Figure 3-14 in which the shaded area depicts the possible costs from any combination of systems, and the heavy line represents the described selected system It is clearly evident that the actual number of data points in the final configurations can vary between 60 and 180 measurements, while the selected concept remains at the optimum point in the cost relationship

3 2 1 3 Other Electromechanical Subsystems - Similar analogies and trades were investigated as required for the remaining astrionics subsystems Inputs from the preliminary aerodynamics flight mechanics/G&C analyses of slosh, bending modes, etc , will determine requirements for and location of body mounted accelerometers and rate gyros

Ordnance devices for vehicle destruct, stage separation, retrograde thrust, and propellant settling were categorically defined to the extent required for determining interfaces, cost, and weight

The range safety/secure command system should not contain redundant receivers and two separate batteries in each stage module, as appear in most current launch vehicles Reliability can be achieved by interlacing single destruct batteries with basic power source and "backing up" the ground command link with inadvertent separation logic if required The inadvertant separation logic would provide an automatic time delayed destruct signal to any major stage segment that separated without a specific command Space and pyrotechnic interface will be provided so that for initial R&D test flights, a redundant receiver/battery combination can be installed to increase confidence but in the operational configuration, in keeping with the emphasis on low cost, it is important to eliminate "emotionally dictated" system redundancy, even if it requires reorienting launch area thinking Modules and interface for the reduced range safety system were included in the system design

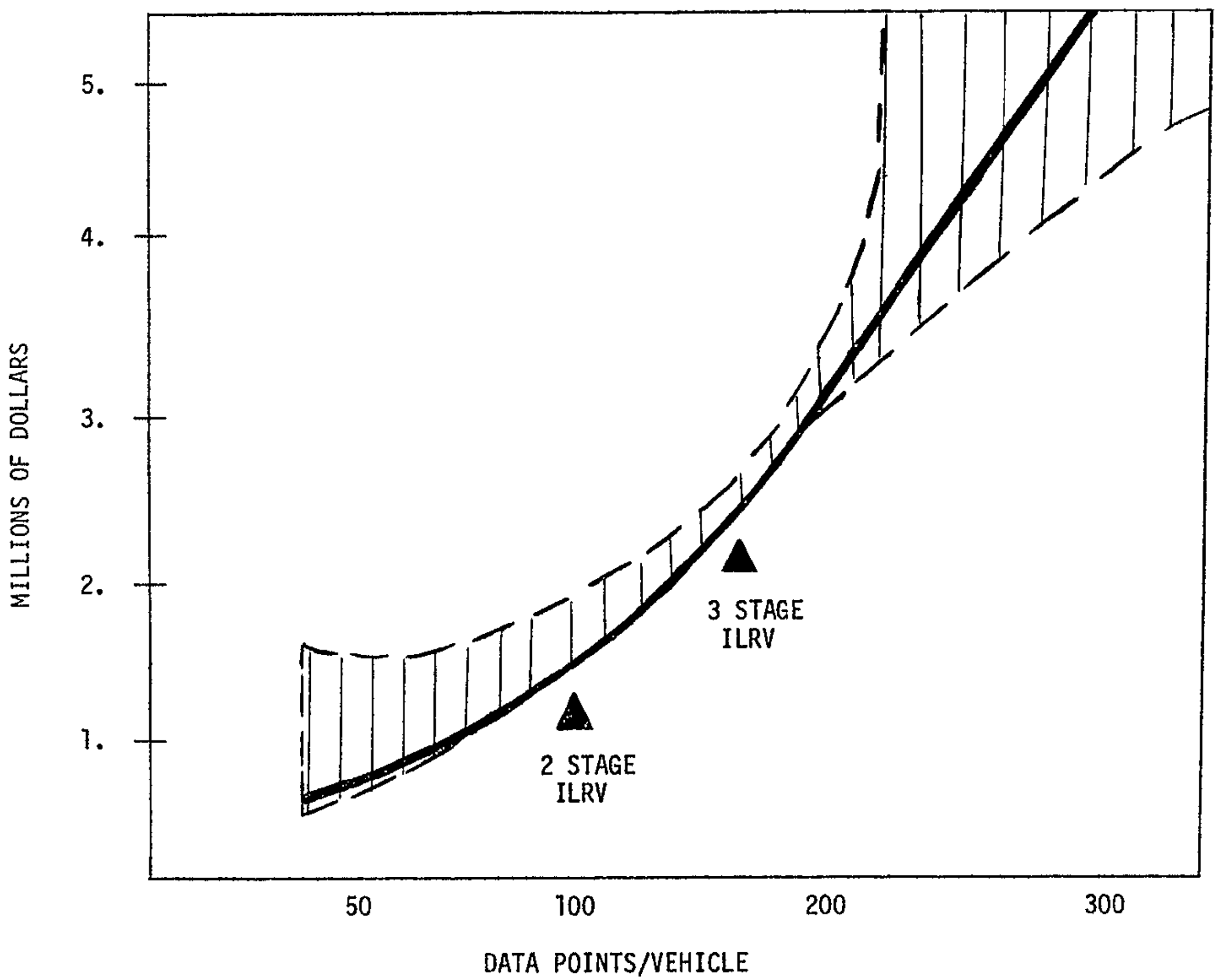


Figure 3-14 Total Instrumentation Costs Including GND Support

Navigation aids such as a C-band transponder were integrated if the functional requirement for them existed.

A general block diagram of the final conceptual electromechanical systems for a two stage vehicle shown in Figure 3-15.

When compared to structures of propulsion systems, astrionics systems reflect some unique relationships with respect to recurring costs The key to designing low-cost systems lies in recognizing hidden elements which generate the greatest costs and reducing or eliminating the causes MDAC internally funded studies considered historical system data from similar contractor and government launch systems designs to isolate the major cost items in astrionics systems Experience indicates that nearly all of these costs are directly related to labor, not to hardware expenditures Areas in which labor costs are prevalent are wire-harness fabrication, system installation, calibration, factory-to-launch-area checkout, and data analysis.

3 2.1 4 Launch Vehicle Electrical Interfaces - Modern technology is continually advancing in automation to reduce labor costs. These advancements are generally applied to mass-produced components and are seldom employed in low production rates associated with space vehicles. Some systems however, could be adopted to more automation by proper exploitation Interconnecting harness fabrication is a prime example The labor cost associated with fabricating, verifying, and installing a large wire harness far exceeds the material costs Figure 3-16 shows the relationship between purchased parts and other labor related costs for a typical space vehicle instrumentation system interconnecting harness MDAC has conducted extensive contracted studies in the area of cost effective electrical wire harness and connectors Data has been derived and documented offering conclusive proof that in new vehicle design and construction flat-cable application can reduce the cost of harnesses by up to 80% when compared to conventional wire systems. This data was evolved in the contracted efforts listed in Table 3-6

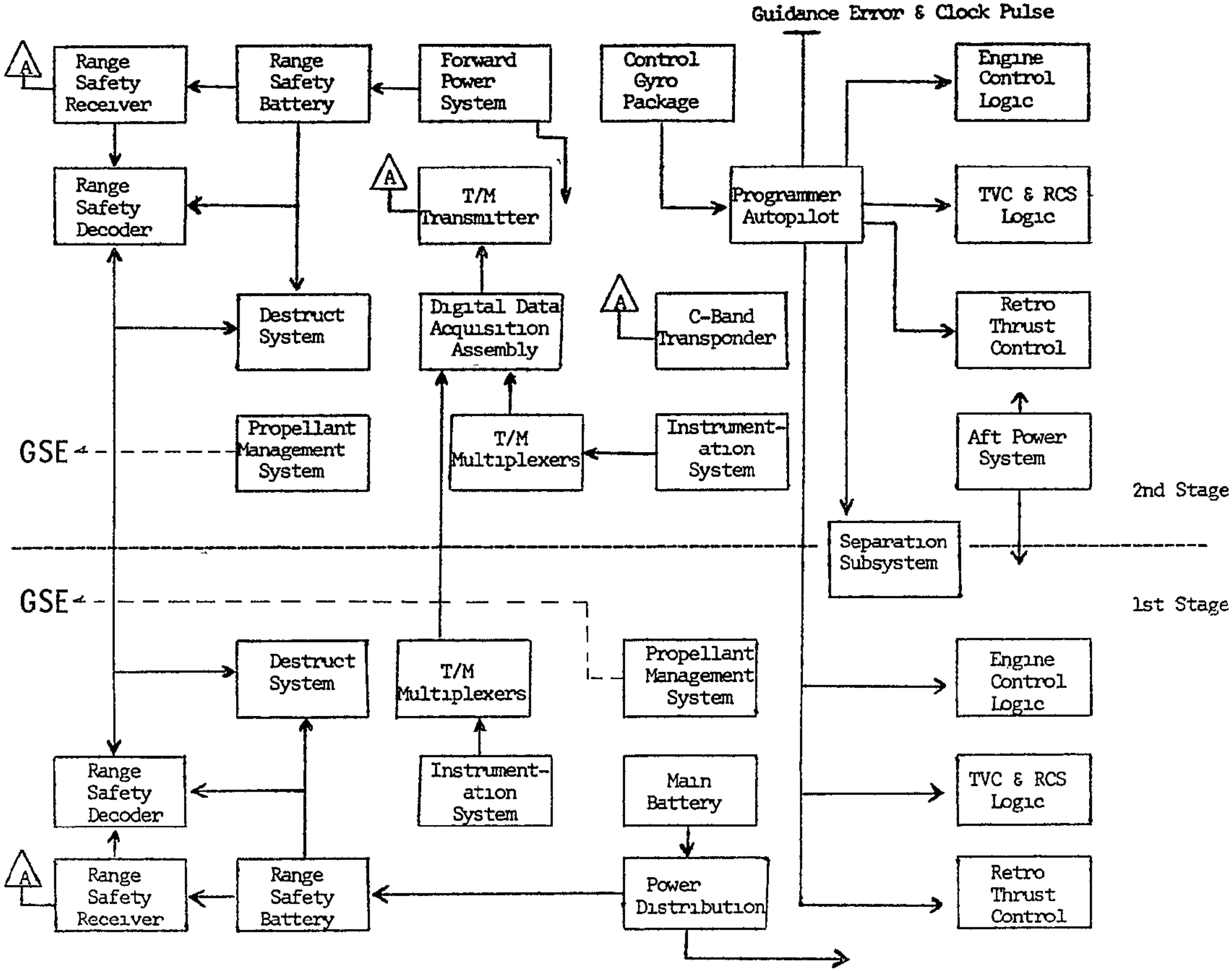
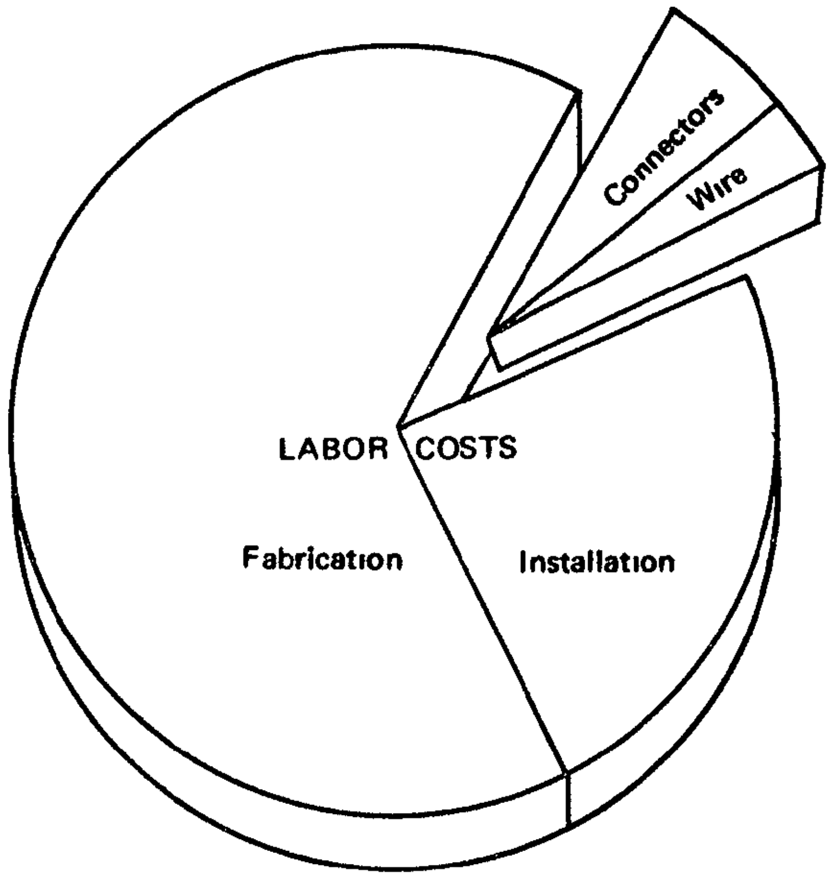


Figure 3-15 Conceptual Electromechanical System



Based on an actual harness containing 523 wires and 144 connectors
Total Cost - $30,000

Figure 3-16 Relationship of Parts to Labor for a Typical, Conventional-Wire Harness

Table 3-6

FLAT-CABLE CONTRACTED EFFORTS

Study Title	Contract Value ($ Thousands)	Government Agency	Study Period	Study Final Reports
Flat-Cable Development	214	NASA MSFC	Jan 65 May 66	SM-53175 - May 66 Vol I and II
Flat-Cable Applications	176	NASA MSFC	Feb 66 Sept 66	DAC-56440 - Sept 66, Vol I and II
Flat-Cable Utilization (System Engineering)	320	NASA MSFC	Mar 67 June 68	DAC-56659 - July 68, Vol I and II, A Flat-Cable Handbook, and 2 MIL specs

During this study, the data on cost effective interconnecting harness utilization was employed to guide system philosophy derivation such that the final astrionics system designs reflect the optimum approach to electrical interface The harness system selected was a complete flat-cable network

When this concept was applied to the basic system layout depicted in Figure 3-15, the tabulation showed a direct savings of 60% in the recurring harness costs and a 35% weight advantage when compared to a conventional wire approach The disadvantage in the inherent lack of modification flexibility associated with flat cable will not be significant in an operational design The cost areas in which the savings will occur are tabulated in Figure 3-17.

3 2 1 5 Astrionics Environmental Protection - The entire thermal energy dissipating portions of the electromechanical systems will be mounted on passive heat sinks incorporated as the structural attach points The maximum flight operation time during the boost phase will be less than 600 sec, therefore, it can be parametrically demonstrated that active thermal conditioning is not required for the electromechanical systems if reasonable consideration is given to module location However, a much more formidable problem exists during the long prelaunch checkout and countdown It will be necessary to provide a ground mounted air conditioning system ducted to the equipment areas in the forward and aft skirt to provide temperature control prior to launch

3 2 2 Generation of Parametric Weight and Costing Data - The second half of the study subtask objective was, having developed a cost effective astrionics system design concept, to apply this approach to the candidate vehicles and size the system for weight and costing data

Previous experience has shown that for any specific vehicle within the medium-to-large size range, an increase in propellant mass by an order of magnitude has very little effect on the weights, cost, and complexity of the electromechanical systems actually required The small increase in system weight brought on by increased interconnecting harness lengths is offset by the latitude of freedom of location within a larger silhouette In practice, however, a huge stage often engenders unnecessary redundancy and instrumentation in the design.

It was decided that to simplify analyses of electro-mechanical systems weight and cost for 28 varigated vehicle configurations, a grouping function was

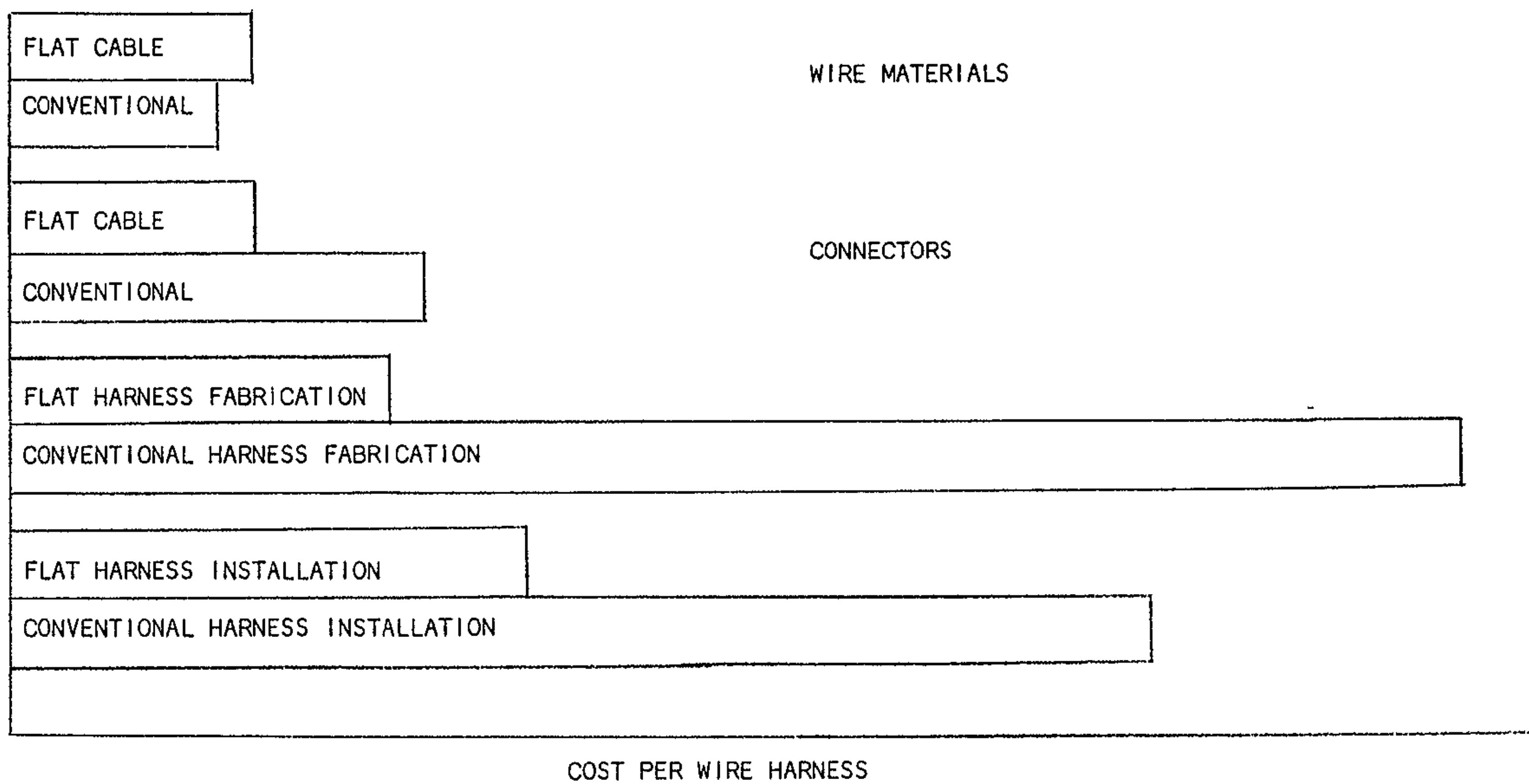


Figure 3-17 Cost Per Wire Harness

required reducing the number of candidates Based on historical data equating astrionics systems to vehicle configurations, a compaction to seven subgroups appeared justifiable The subgroups were as follows

- Stages of a two-stage vehicle
  - I With similar engine systems (quantity and TVC)
  - II With different engine systems
  - III With cryogenic engine systems
- First and second stages of a three-stage vehicle
  - IV With similar engine systems
  - V With different engine systems
  - VI With cryogenic engine systems
- VII Third Stages

The analysis began with Subgroup I vehicles, described as two-stage boost vehicles with similar noncryogenic engine systems The similarity is reflected in the same type of engines and the same thrust vectoring system on each stage ILRV candidate vehicles that fall into this classification are

1 Storable, pressure first stage, storable pressure second stage
2 LO_2-RP, pressure first stage, LO_2-RP, pressure second stage
3 LO_2-RP, pressure first stage, LO_2-RP, second stage
4 LO_2-RP, pump first stage, LO_2-RP, pump second stage
5 Solid 260 first-stage, Solid 260 second stage
6 Storable, pump first stage, storable, pump second stage

3 2 2 1 Weight Inputs to Performance Analysis - The basic system layout pictured in Figure 3-15 was sized to the subject vehicle configurations and subsystem weights determined to the major module level Weights were derived by first selecting available units either "off-the-shelf" or on other vehicle programs which were functionally adequate and exhibited a high order of cost effectivity when compared to other similar devices If a practical feasible unit was not readily available, current astrionics and conceptual vehicle studies were examined for likely candidates under development As a last resort, an optimum device was hypothesized within the technology base expected for the mid-1970's to the extent required to make weight approximations

Based on a "test case" approach, the a priori weight assignments were tentatively verified to be accurate within 25% for a hypothetical module and 15% for a developmental unit Actual weights were utilized for existing devices In each candidate system examined, the ratio of existing to conceptual modules was

6 1 or greater yielding an overall astrionics weights accuracy of no worse than ± 10%. Tables 3-7 and 3-8 respectively depict the calculated weights for the first and terminal stage of Subgroup I vehicle candidates

When the astrionics weight approximation (2, 000 lb total) is compared to the minimum dry weight for a candidate two-stage vehicle (200, 000 lb) or the lift-off minimum of 2 million lb, it can be seen that the astrionics represents less than 1% of the dry weight and 0 1% of the launched weight Thus the 15% tolerance on astrionics weight approximations becomes allowable.

3.2.2.2 Cost Inputs to pricing Analysis - The astrionics modules were described for costing by calling out actual modules by nomenclature or part number or by relating conceptual devices to percent complexity of existing 5 star program modules. Pricing analysts then used this data to generate actual cost values Again based on "test-case-spot-check" verification techniques, quantitative accuracy for astrionics pricing was assessed to be within 10% of true recurring cost On Subgroup I vehicles, the astrionics cost relative to the total vehicle cost varied from 10% to 25%, with the average and median both below 20% Therefore, the cost error induced by astrionics pricing techniques was generally less than 2%

3 2 2 3 Astronics Variations Between Vehicle Subgroupings - As the analysis broadened to include more vehicle configurations categorized in the original Subgroups II thru VII, it became increasingly evident that perturbations to total vehicle weight and cost due to variations in the astrionics systems were relatively insignificant The largest variations occurred when cryogenic fuels were employed in the stage For example, in avionics systems of the first stage for two and three stage all cryogenic vehicles, the avionics costs and weights will be similar if not identical to LO_2/RP first stage systems with the following exceptions

- Engine control logic will increase in weight by 5 lb and in cost by 10%
- Signal conditioning racks will increase in weight by 8 lb and in cost by 15%
- Fifteen additional measurements increases sensor weight by 36 lb and cost by 25%.
- Propellant management logic (loading, venting) increases in weight by 10 lb and doubles in cost.
- Interconnecting harness weight is increased by 40 lb

Table 3-7

FIRST STAGE SUBSYSTEMS

Guidance and Control		57 lb
Engine Control Logic	(23)	
TVC and RCS Logic	(26)	
Retrograde Thrust Control	(8)	
Power and Distribution		106
Main Battery	(60)	
Logic and Measurement Power Supplies	(20)	
Power Distributer	(26)	
Instrumentation		138
Submultiplexer	(12)	
Signal Conditioning Racks	(30)	
40 Measurements (Sensor to Mux)	(96)	
Destruct/Range Safety		48
Range Safety Receiver	(7)	
Decoder	(9)	
Antenna	(7)	
Safe and Arm	(3)	
Shaped Charge	(9)	
Range Safety Battery	(13)	
Separation Subsystem		25
Propellant Management		30
Interconnecting Harness		225
Total Modules		629
Three Passive Ambient Plates		36
Mounting and Attach Hardware		157
Total First Stage Electromechanical		822 lb

NOTE It is general practice to include the retrograde motorcases as part of electromechanical systems However, for this exercise, where stage weight and thrust are not a point design, the motorcases are included as a formula factor with the retrograde propellant and equated to required retro thrust

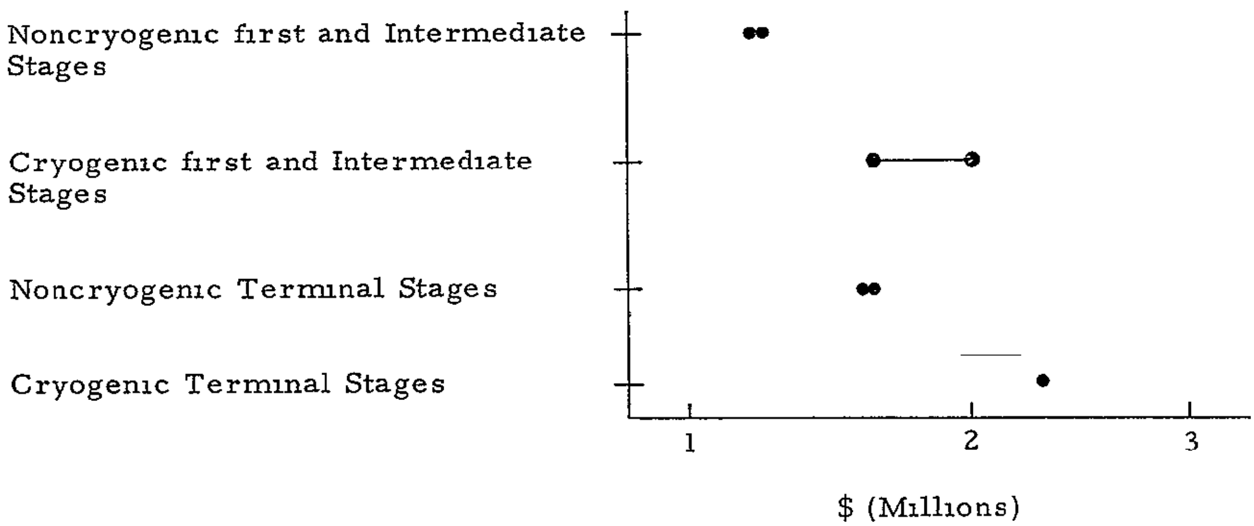
Table 3-8

TERMINAL STAGE SUBSYSTEMS

Guidance and Control		125 lb
Programmer/Autopilot	(45)	
Control Gyro Package	(17)	
Engine Control Logic	(23)	
TVC and RCS Logic	(26)	
Retrograde Thrust Control	(8)	
C-band Transponder and Antenna	(6)	
Power and Distribution		182
Main Batteries	(120)	
Logic and Measurement Power Suppl	(26)	
Power Distribution	(36)	
Instrumentation/TM		263
FM/FM Transmitter	(15)	
Antenna System	(8)	
ODAA	(24)	
Multiplexer	(20)	
Submultiplexer	(12)	
Signal Conditioners Racks	(40)	
60 Measurements (Sensor to Mux)	(144)	
Destruct/Range Safety		48
Separation Subsystem		25
Propellant Management		30
Interconnecting Harness		210
Total Modules		883
Five Passive Ambient Plates		60
Mounting and Attach Hardware		200
Total Second Stage Electromechanical		1,143 lb

The astrionics weight variations between the two stages discussed in the example is 99 lb or about 10% of total stage astrionics weight and 0 05% of the total vehicle dry weight (10^2 2×10^5) This analogy was researched and found to hold true for all stages of all candidates. Therefore, three quantitative average weights for first, intermediate (second stage of a three-stage configuration), and terminal stages were determined and utilized as constants for performance analyses

The problem was slightly more complex with respect to the pricing data, where delta (Δ) cost was greater and astrionics incurred a more significant percentage of vehicle total costs However, again the largest variations were apparent when cryogenic and noncryogenic stages were compared A general comparison of total astrionics cost is shown below



Astrionics costs relative to total vehicle cost varied between 10% and 40%, with the average and median for noncryogenic stages below 20% (in almost all noncryogenic configurations astrionics were below 25% of total vehicle cost) and cryogenic stages astrionics averaging 30% of total vehicle cost

The aforestated 10% accuracy for astrionics pricing analysis was determined for noncryogenic stage systems, which included a larger proportion of conceptual versus existing modules (approximately 1 6) In the cases involving cryogenic stages, more defined subsystem modules and precise data were available from MDAC's close contact with low cost or improved versions of Saturn hardware and the accuracy of cost figures was better than ±10% Thus, even when the relationship of astrionics to total vehicle costs reach 40% the increased accuracy of pricing kept the astrionics induced errors to less than 5% of the total vehicle configuration costs

3 2 3 Summary of Electromechanical System Subtask - Electromechanical system subtasks are summarized as follows

- A low cost launch vehicle must have an "operational" configuration astrionics system in the initial design and utilize kits for "add-on" R&D measurements, etc
- Guidance error signals and mission real time clock sequencing should be obtained from the payload and format translated in the launch vehicle
- Engineering design techniques must include cost effectivity as a principal criterion
- Utilizing a flat cable harness system will yield large savings in both cost and weight for all vehicle stages
- System redundancy requirements must be real and not emotionally dictated
- Astrionics cost and weight can be made relatively insensitive to launch vehicle configuration thereby reducing its importance as a selection factor (Exceptions - cryogenic or "exotic" fuels, increasing engine quantity, multiple parallel tanks, dissimilar stage TVC systems, and active closed-loop propellant management all engender increases in astrionics complexity/cost),
- An astrionics package costing less than 20% of the total vehicle cost is readily achievable for the ILRV configuration selected

3 3 Structure System Definition - For the extensive matrix of candidate propellants and the possible stack-up arrangements, it was necessary to establish a common base from which valid structural comparisons could be made To accomplish this, a representative propellant load was selected and a structural arrangement "peg-point" design was drawn for each stage and each propellant

Next, a loading analysis technique was developed that permitted determination of all primary structural component materials, sizes, and type of construction on a consistent, comparative basis Weights derived from these analyses were theoretical minimum and did not account for joints, discontinuities, access, separation, component support, etc Weights for these are accounted for in the weight analysis (Section 4.1) These material, construction, and weight data were then supplied to the parameteric cost and weight exercises.

The following sections define the ground rules established to assure comparative consistency throughout the study in determination of weights for tankage, cases, skirts, interstages, and thrust structure Following these ground rules and working curves are the 32 configurations with applicable data developed during this part of the study

3 3 1 Configuration - Ground rules established to define the configurations are in three basic categories, pump-fed, pressure-fed, and SRM as summarized in Table 3-9 Stage diameters selected are similar to current hardware and studies, or logical intermediate points for parametric sizing The propellant tank ullage factors used are based on Saturn technology, and hemispherical end domes were utilized on all tanks for simplicity The thrust structure included-angle is shown, where applicable, and is tangent to the aft tank dome This also follows Saturn S-IVB design

Table 3-9
CONFIGURATION GROUND RULES

Items	Pump-fed	Pressure-fed	SRM
• Stage diameters considered	120, 180, 216, 260, 300, 396		260
• Propellant tank ullage factor	1 05	1 05	NA
• Tankage domes	Hemispherical		
• Thrust structure included angle	90°	90°	NA
• Tank/tank clearance	Figure 3-15		NA
• Interstage lengths	Figure 3-16		From propulsion input

Figure 3-15A shows the clearance between the oxidizer and fuel tanks versus stage diameter The clearance increases with stage diameter to allow for access to and installation of larger propulsion lines and components Approximate inter-stage lengths were developed from Figure 3-16A, which allows for estimated engine length, tank diameter, thrust structure length and engine/lower stage forward dome clearance

3 3 2 Loads Criteria - The ground rules used to obtain vehicle loading are common for the pump-fed, pressure-fed and solid systems, except as noted

A manned factor of safety of 1 4 was used for all structural component sizing, except for the SRM case, where a 1 25 factor on MEOP was applied This is standard for all SRM manufacturers and was used on previous MDAC studies

The weight of a typical re-entry vehicle forward of the terminal stage, was assumed in the loading calculations Using the two-stage, pump-fed, LO_2-RP vehicle as a test configuration, shell loading intensity was determined for maximum acceleration, for maximum αq, and for ground hold were appropriate. For this analysis, a stage λ' of 0 90 was used to estimate the total structural (and engine) weights from given stage propellant loads. This weight was distributed uniformly over the stage length, and propellant weights were assumed to be introduced into the shell at the aft dome/skirt joint

Maximum "g" - A maximum acceleration of 6 g's (limit) was used with 100 percent depletion of first stage propellant to determine loading for this flight condition Next, a maximum bending distribution curve, as shown in Figure 3-17A, was developed from a previous study for a vehicle of similar geometry, mission, and payload

Maximum Moment - An acceleration level of 2 g's was assumed to combine with the bending (per reference study) at 70 sec into flight, assuming 42 percent first stage propellant depletion for this flight mode

Ground Hold - Ground wind bending loads are also shown.

When sidewall loading intensities were compared for the indicated conditions, it was found that the αq condition was critical for structure forward of the first stage. First stage loading was approximately equal for the two conditions Because it is far more convenient to determine loading intensity as a function of g-level than it is to plot each different vehicle configuration against the bending

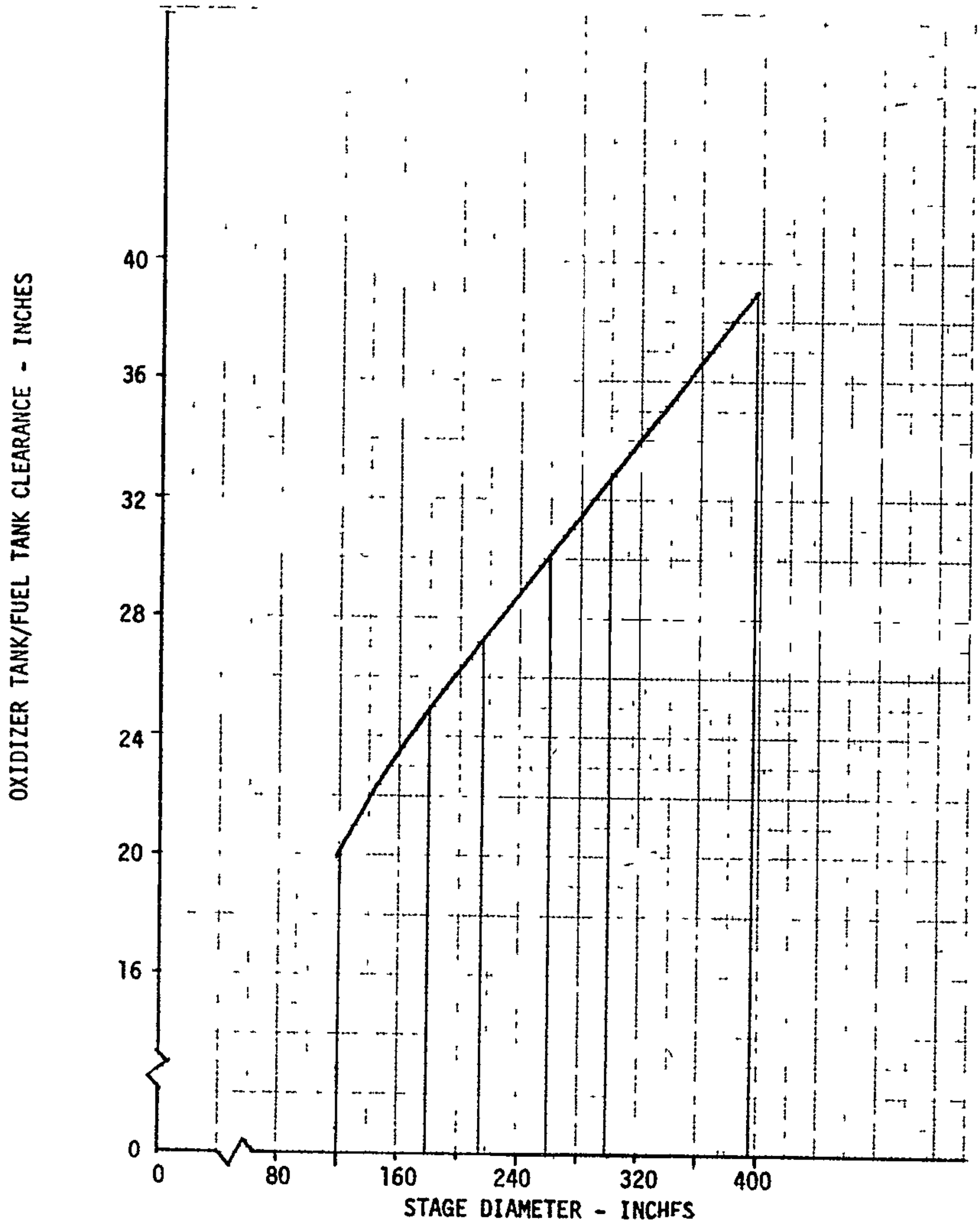


Figure 3-15A Tank-To-Tank Clearance Used for Stage Stack-Height Determination

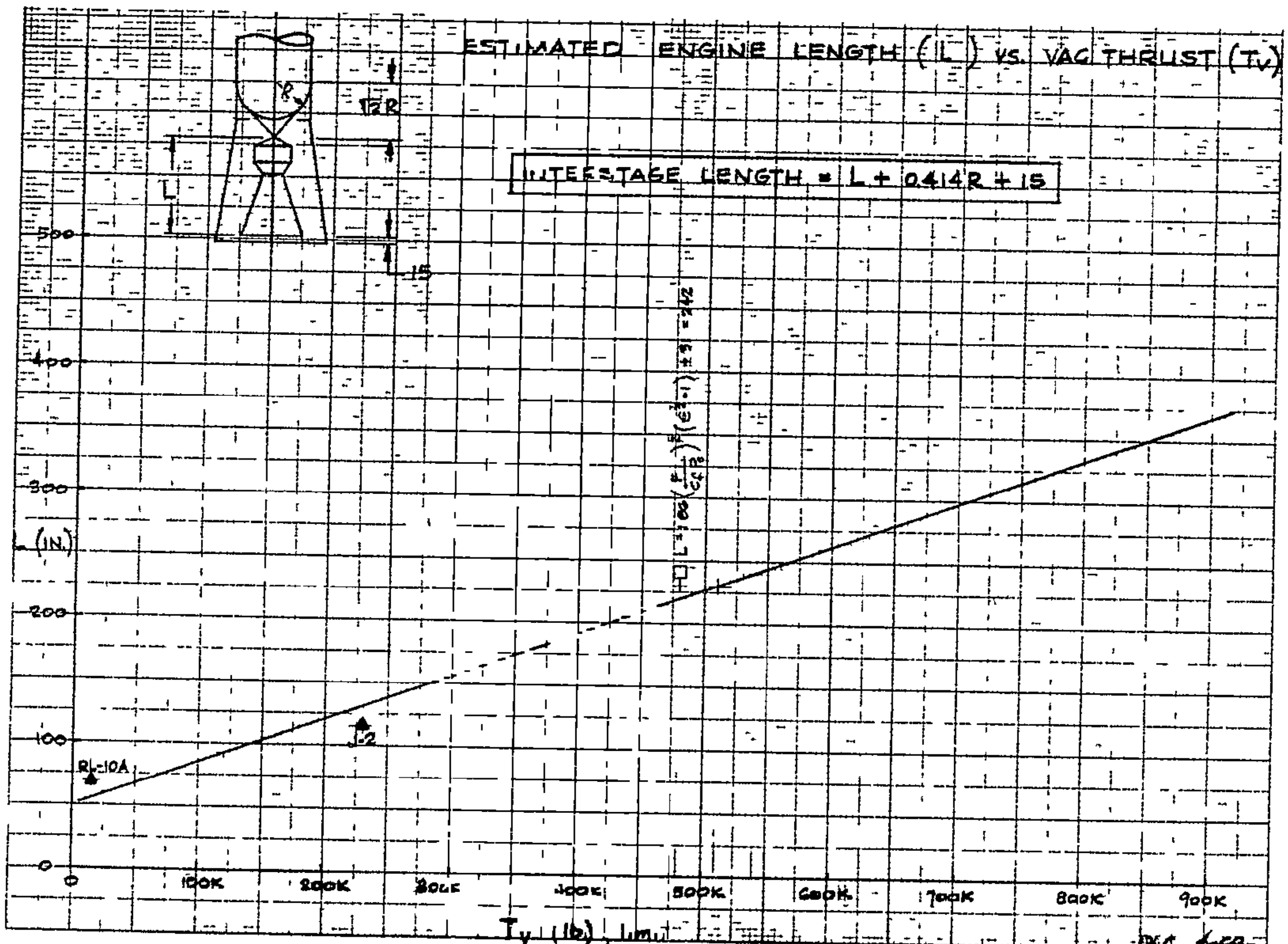


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Figure 3-16A Interstage Length Vs Engine Thrust

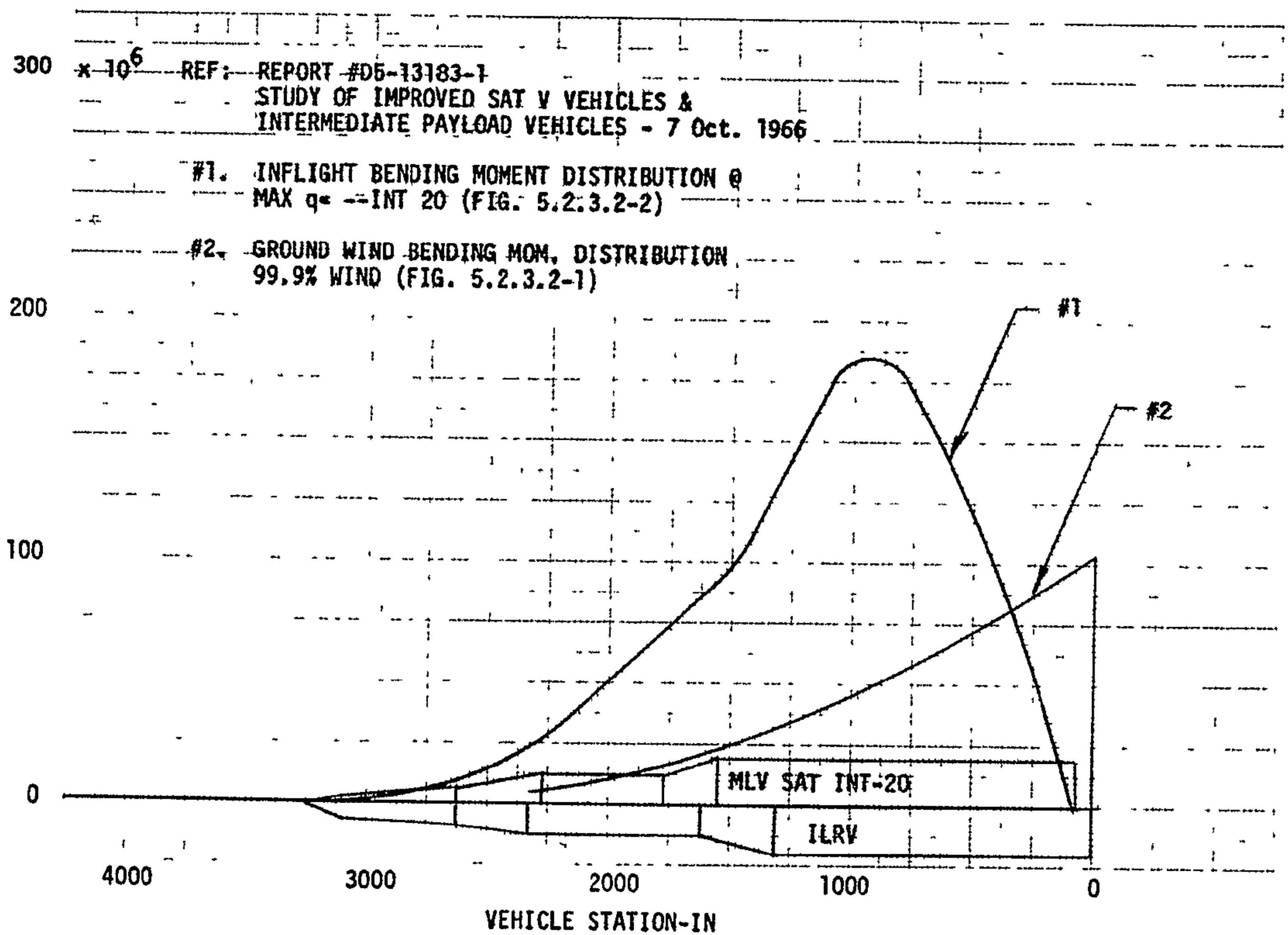


Figure 3-17A Vehicle Bending Moment Distribution moment distribution curve, an equivalent g-level was, therefore, determined for each stage component These were as follows

First stage		6 g constant
Second stage	aft skirt,	6g
	center skirt,	9 g
	forward skirt,	12 g
Third stage		12 g constant

On the pump-fed vehicles, tankage sidewall was found critical for the unpressurized, unfueled, ground wind bending condition The sidewalls on the pressure-fed and solid vehicles are sized by the tank internal pressure conditions

3 3 3 Tankage Design - The tankage design parameters for the pump-fed, pressure-fed and SRM systems are summarized in Table 3-10 Aluminum was considered for the pump and pressure-fed vehicles and maraging steel for the pressure-fed and solid vehicles Material selections, based on cost-effectiveness, allowables, and densities used are tabulated

Forward and aft domes shape, type of construction, and critical design conditions are shown The same data is also shown for the cylindrical sidewalls A 45° waffle stiffening pattern, similar to the S-IVB stage, is required on the pump-fed system to withstand the ground-wind compression loading This design curve is shown in Figure 3-18 The pressure-fed system requires only monocoque sidewalls since the internal pressure loading is critical Monocoque sidewalls and extensions are also acceptable on the SRM stages In this case, the critical pressures and ground wind loads require nearly the same wall thickness which is near optimum The limit internal pressures are also indicated for all three systems

3 3 4 Thrust Structure Design - The thrust structure design is applicable only for the pump and pressure-fed systems It is a 90° included angle tangent and attaches to the aft dome Thrust structure loading was defined by propulsion as a function of stage propellant (W_p) First and upper stage thrust loads were equal to 1 6 x stage W_p and 1 2 x stage W_p, respectively With an engine thrust level defined, a design curve was generated to obtain thrust structure component weight as shown in Figure 3-19 This is based on external stringer/skin/internal frame construction made from 7075-T6 aluminum sheet and extrusion

3 3 5 Skirt and Cylindrical Interstage Design - All the cylindrical skirts and interstages have the same type of integrally machined skin and stringer construction A recent company study has indicated this is a lower-cost concept than conventional skin-stringer construction for approximately the same weight Both

Table 3-10
TANKAGE DESIGN SUMMARY

Items		Pump-Fed	Pressure-Fed	SRM
• Tankage Material		2021-T81 Alum		18 Ni-250 Mar Steel
• Ultimate Tensile Allowable, psi		66,000		250,000
• Material Density, lb/in^3		0 100		283
• Forward Dome	- Shape	Hemispherical		
	- Construction	Monocoque		
	- Design Condition	Pressure		
• Aft Domes	- Shape	Hemispherical		
	- Construction	Monocoque		
	- Design Condition	Pressure + 1 25 Liftoff g Prop Head		
• Sidewall	- Construction	Waffle	Monocoque	Monocoque
	- Design Condition	Ground Wind	Pressure	Pressure
	- Extensions	NA	NA	Monocoque
• Limit Internal Pressures, psia		All Stages, 50	Stages I, II, 450 Stage III, 300	MEOP = 672

types utilize the same internal stabilizing frames made from 7075-T6 extrusion The integral skin-stringer panels are machined from 7075-T6 plate

A design curve (Figure 3-20) was then developed to determine unit structural weight versus loading intensity for these components This curve is based on current S-IVB hardware and various modified S-IVB studies and also lists the design assumptions used

3 3 6 Conical Interstage Design - Conical interstages are used between two stages of different diameter Conventional skin-stringer was selected for this design because of the difficulty of machining a conical section The same extruded internal frames are used and all material is 7075-T6 aluminum sheet or extrusion The design curve from Figure 3-20 was used to obtain component weight

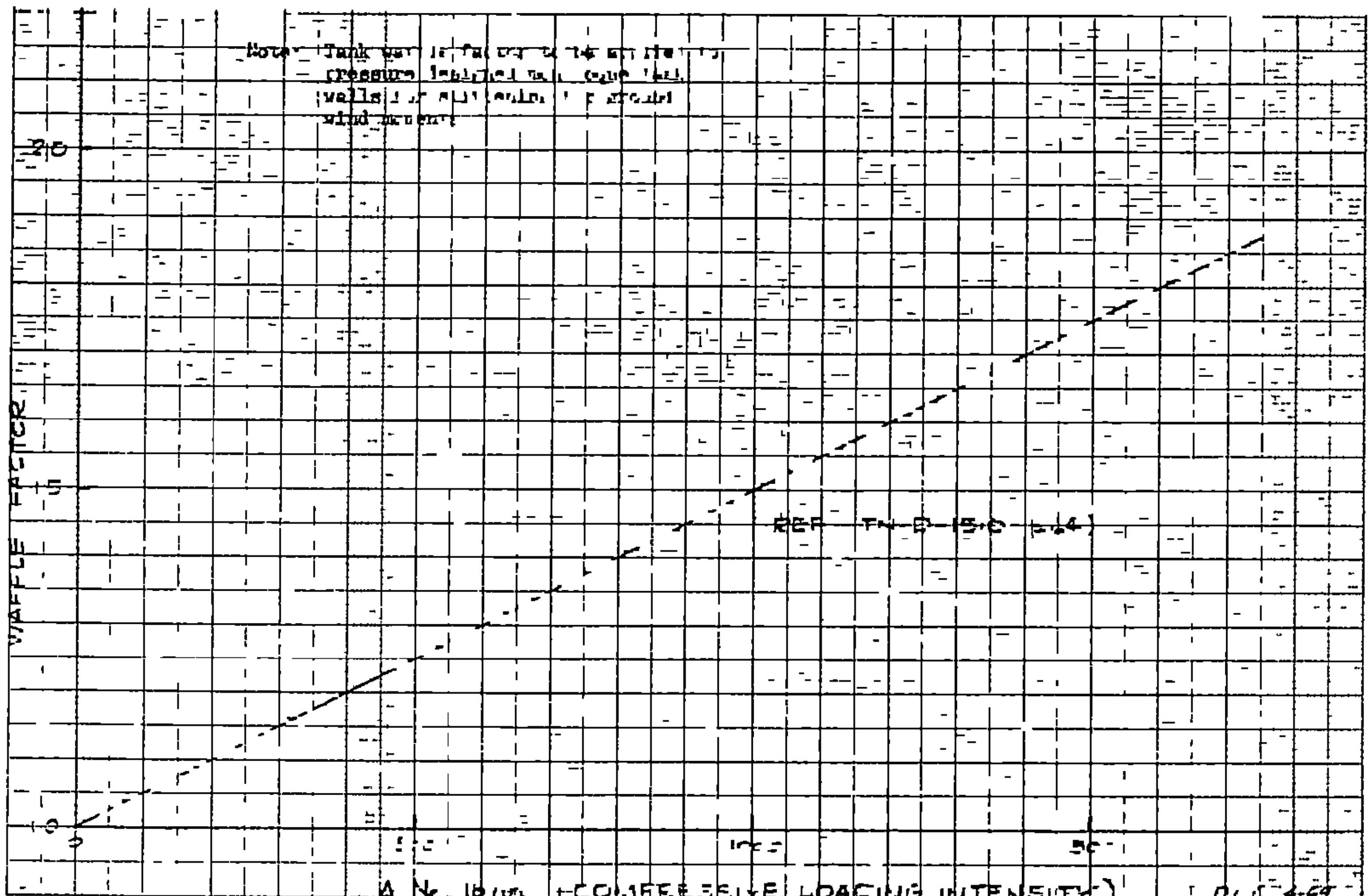


Figure 3-18 Tank Waffle Factor Vs Loading Intensity

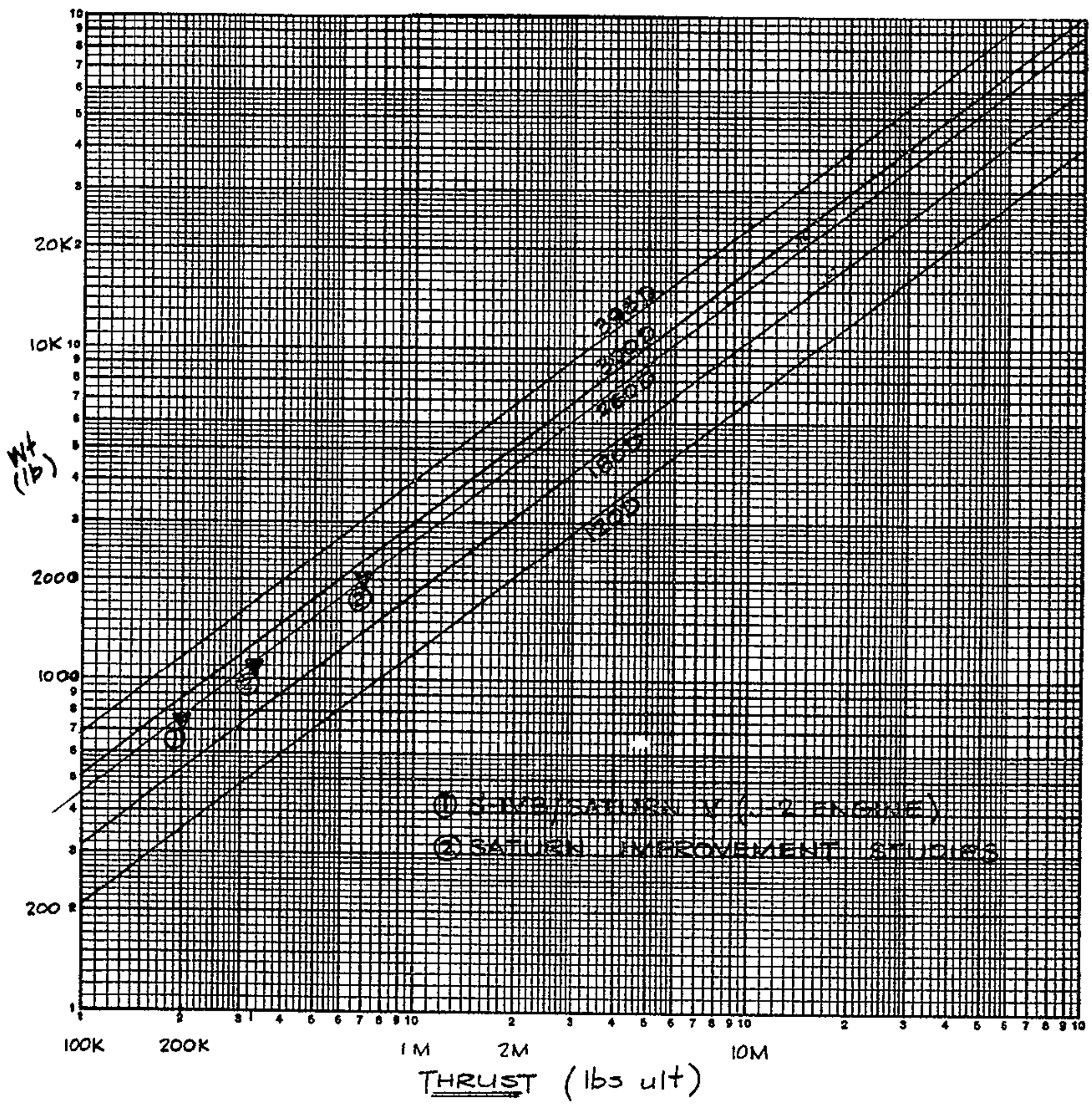


Figure 3-19 Thrust Structure Weight vs Engine Thrust

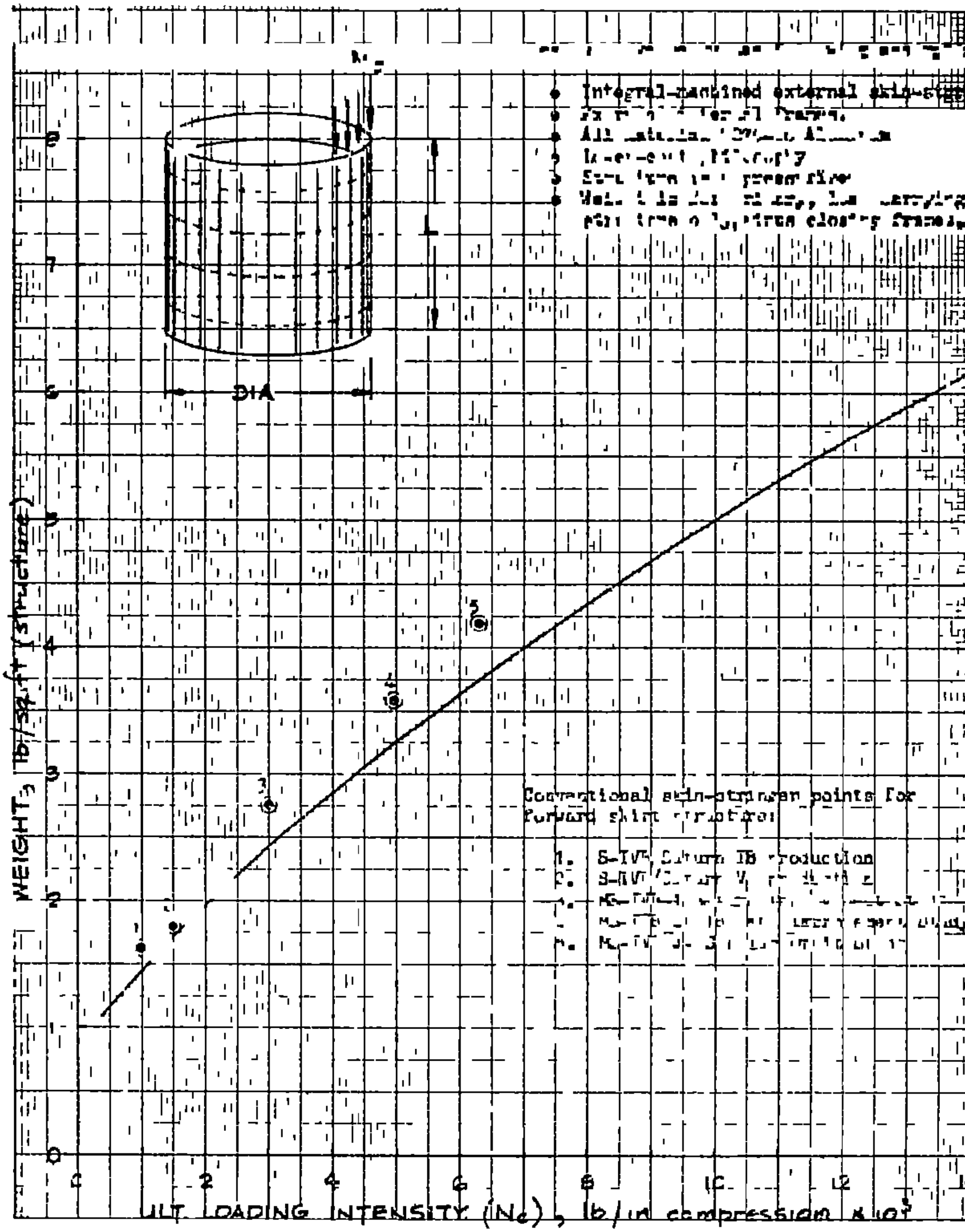


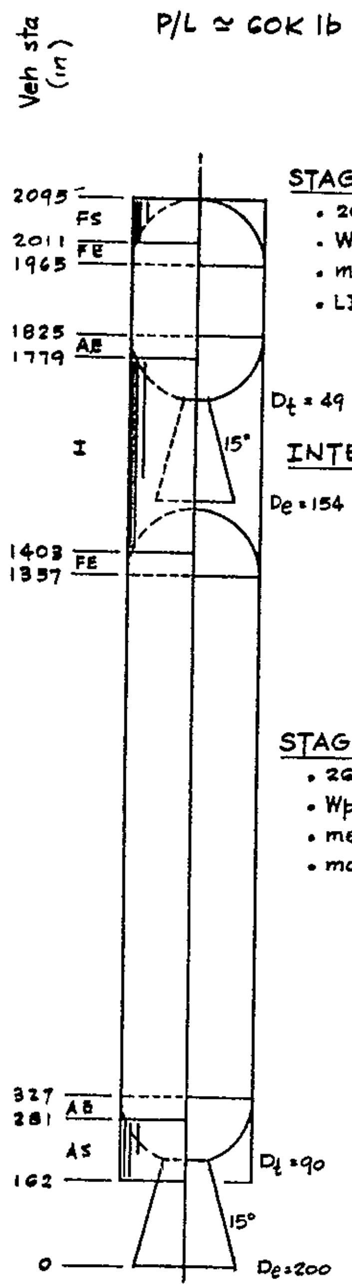
Figure 3-20 Structural Weight vs Loading Intensity

3 3 7 Peg-point Designs - The matrix of vehicles showing the "peg-point" configurations and critical parameters used in determining the weights of the various major structural components is shown in Figures 3-21 through 3-41.

When a given stage has been sized for one launch vehicle, that stage may be used for subsequent stack-ups in combination with different upper and lower stages When this is done with a mating stage(s) of different diameter from the original mating, only the interstage weight need be determined for the new combination

In summary, "peg-point" designs were defined for the candidate vehicles and included details such as stage lengths and diameters, propellant loadings, etc The critical loading conditions are shown from which the component weights were calculated These weights were then used as inputs to the parametric weight and costing analyses

X_1 : SOLID 260



STAGE II
- 260 D SRM
- Wp = 10M lb (0.5-2.0)
- meop = 672 psia
- LITVC noz, ϵ = 100

INTERSTAGE

STAGE I
- 260D SRM
- Wp = 3.4M lb (1.0-8.0)
- meop = 672 psia
- mov noz, ϵ = 3.0

		Wt (lb)*
Forward Skirt		760
Insulated chamber		55,300
• Steel case	29,000	
• Insul & liner	16,400	
• Fwd & aft ext	9800	
• Rings, joints	600	
Nozzle, LITVC		16,280
Igniter motor		1,000
Interstage		11,920
Insulated chamber		157,300
Steel case	123,000	
• Insul & liner	24,500	
• Fwd & aft ext	9,800	
• Rings, joints	600	
Mov nozzle		40,600
Igniter motor		3,400
Aft skirt		4800

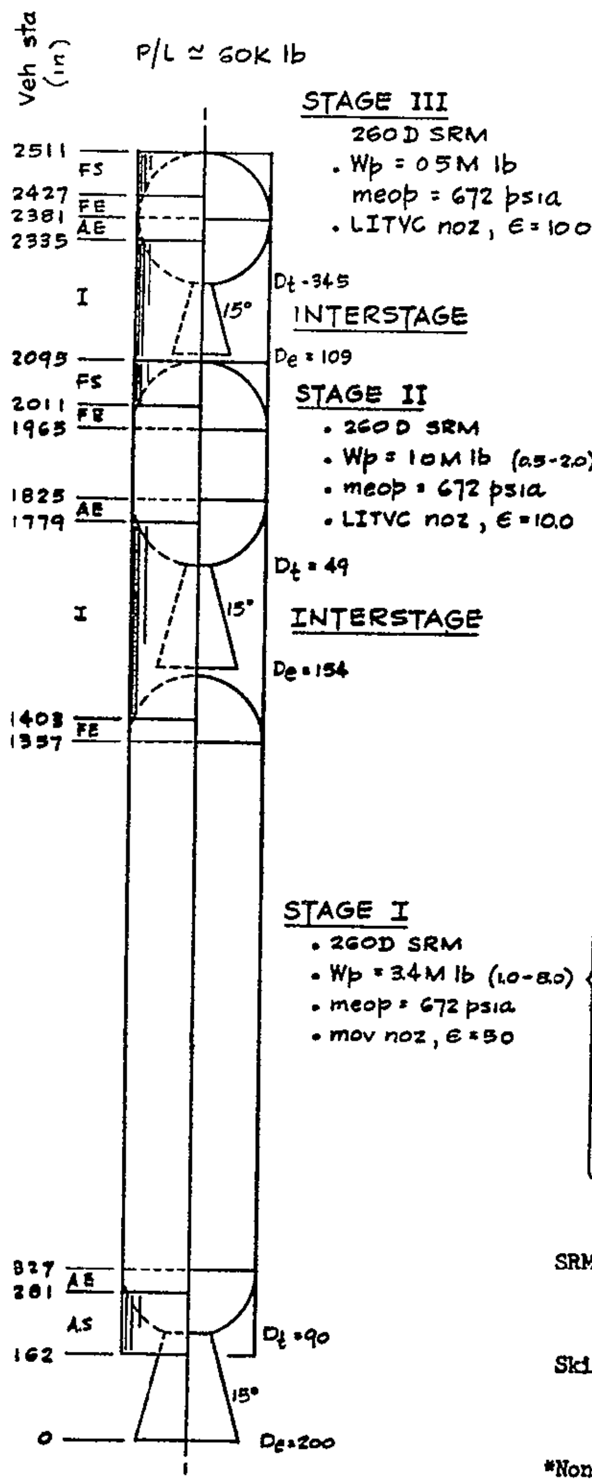
SRM Case: 18Ni-250 mar. steel, F_{tu} = 250 ksi, hemi-spherical, monocoque domes, monocoque sidewalls and extensions.

Skirts and cylindrical interstages: 7075-T6 alum plate, integ machined skin-stringer, internal frames.

*Non-optimum weight factors not included:
1.02 for SRM case & extensions
1.20 for interstages
1.44 for skirts

Figure 3-21

X_2 : SOLID 260



		Wt (lb)* same as
Forward skirt		760 X_1
Insulated chamber		38,500
Steel case	13,000	
• Insul & liner	15,100	
• Fwd & aft ext	9,800	
• Rings, joints	600	
Nozzle, LITVC		
Igniter motor		500
Interstage		5,150
Forward Skirt		1,800
Insulated chamber		55,300 X_1
• Steel case	29,000	
• Insul & liner	16,400	
• Fwd & aft ext	9800	
• Rings, joints	600	
Nozzle, LITVC		16,280 X_1
Igniter motor		1,000 X_1
Interstage		15050
Insulated chamber		157,300 X_1
• Steel case	123,000 ✓	
• Insul & liner	24,500	
• Fwd & aft ext	9,800	
• Rings, joints	600	
Mov nozzle		40,600 X_1
Igniter motor		3,400 X_1
Aft skirt		4800 X_1

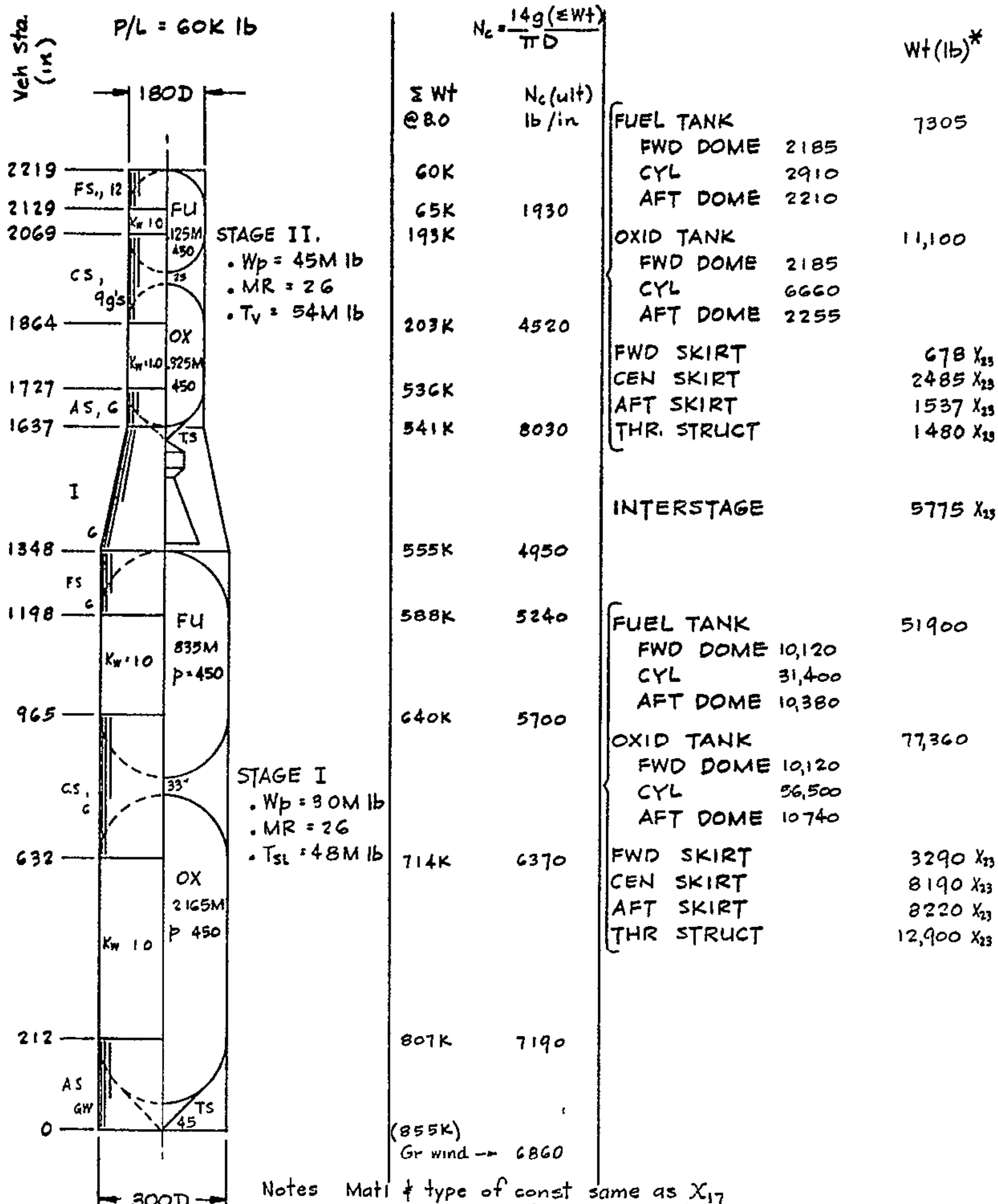
SRM Case: 18Ni-250 mar. steel, F_{tu} = 250 ksi, hemi-spherical, monocoque domes, monocoque sidewalls and extensions.

Skirts and cylindrical interstages: 7075-T6 alum. plate, integ. machined skin-stringer, internal frames.

*Non-optimum weight factors not included:
1.02 for SRM case & extensions
1.20 for interstages
1.44 for skirts

Figure 3-22

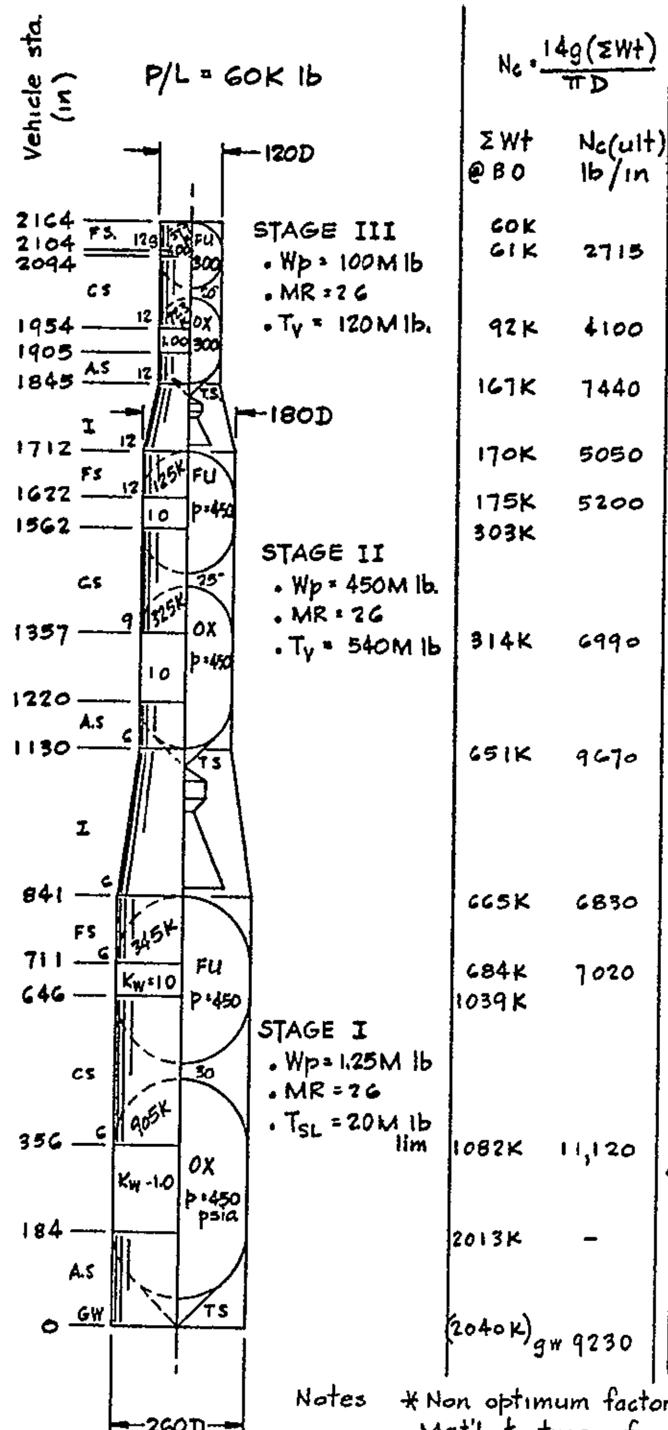
X_{12}: STORABLE, PRESSURE



Notes Mat'l & type of const same as X_{17}
* Non-optimum wt factors not included: 1.20 for interstage; 1.44 for all other components

Figure 3-23

X_{13} : STORABLE, PRESSURE



			Wt (lb)*	
FUEL TANK			1012	
	FWD DOME	432		
	CYL	144		
	AFT DOME	436		
OXID TANK			1577	
	FWD DOME	432		
	CYL	704		
	AFT DOME	441		
FWD SKIRT			361	X_{28}
CTR SKIRT			1060	X_{24}
AFT SKIRT			652	X_{28}
THR STRUCT			310	X_{28}
INTERSTAGE			1620	X_{28}
FUEL TANK			7305	X_{12}
	FWD DOME	2185		
	CYL	2910		
	AFT DOME	2210		
OXID TANK			11,100	X_{12}
	FWD DOME	2185		
	CYL	6660		
	AFT DOME	2255		
FWD SKIRT			1175	X_{24}
CTR SKIRT			3220	X_{24}
AFT SKIRT			1732	X_{24}
THR STRUCT.			1480	X_{23}
INTERSTAGE			6125	X_{24}
FUEL TANK			19825	
	FWD DOME	6580		
	CYL	6575		
	AFT DOME	6670		
OXID TANK			30,770	
	FWD DOME	6580		
	CYL	17,410		
	AFT DOME	6780		
FWD SKIRT			2950	X_{24}
CTR SKIRT			8800	X_{24}
AFT SKIRT			7440	X_{28}
THR STRUCT			5800	X_{24}

Notes * Non optimum factors same as X_{17}
Mat'l & type of const same as X_{17}

Figure 3-24

X_{14}:

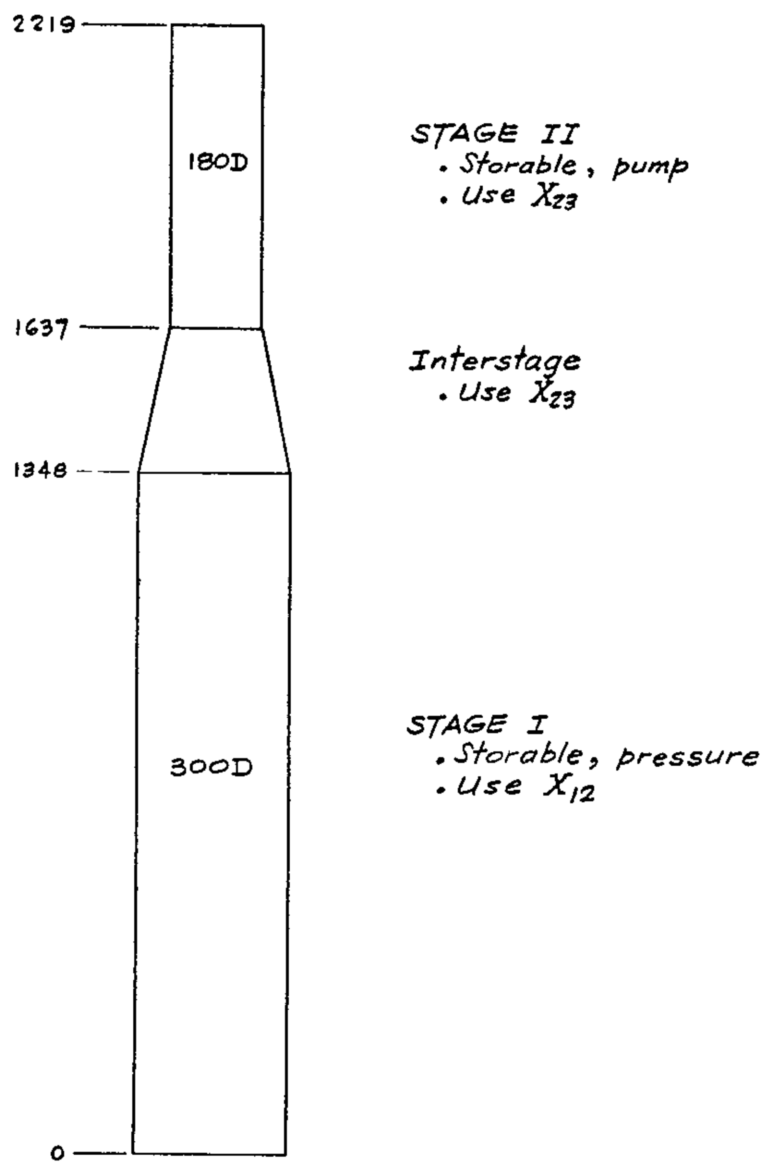


Figure 3-25

X_{15}:

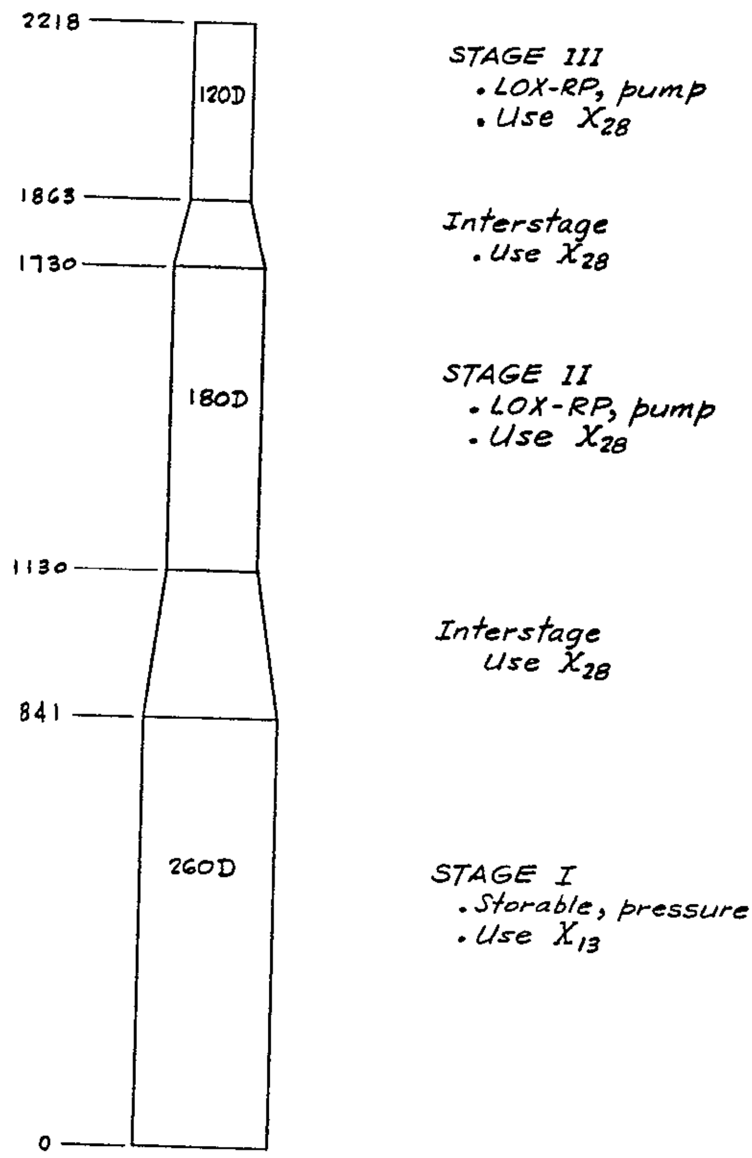


Figure 3-26

X_{16}:

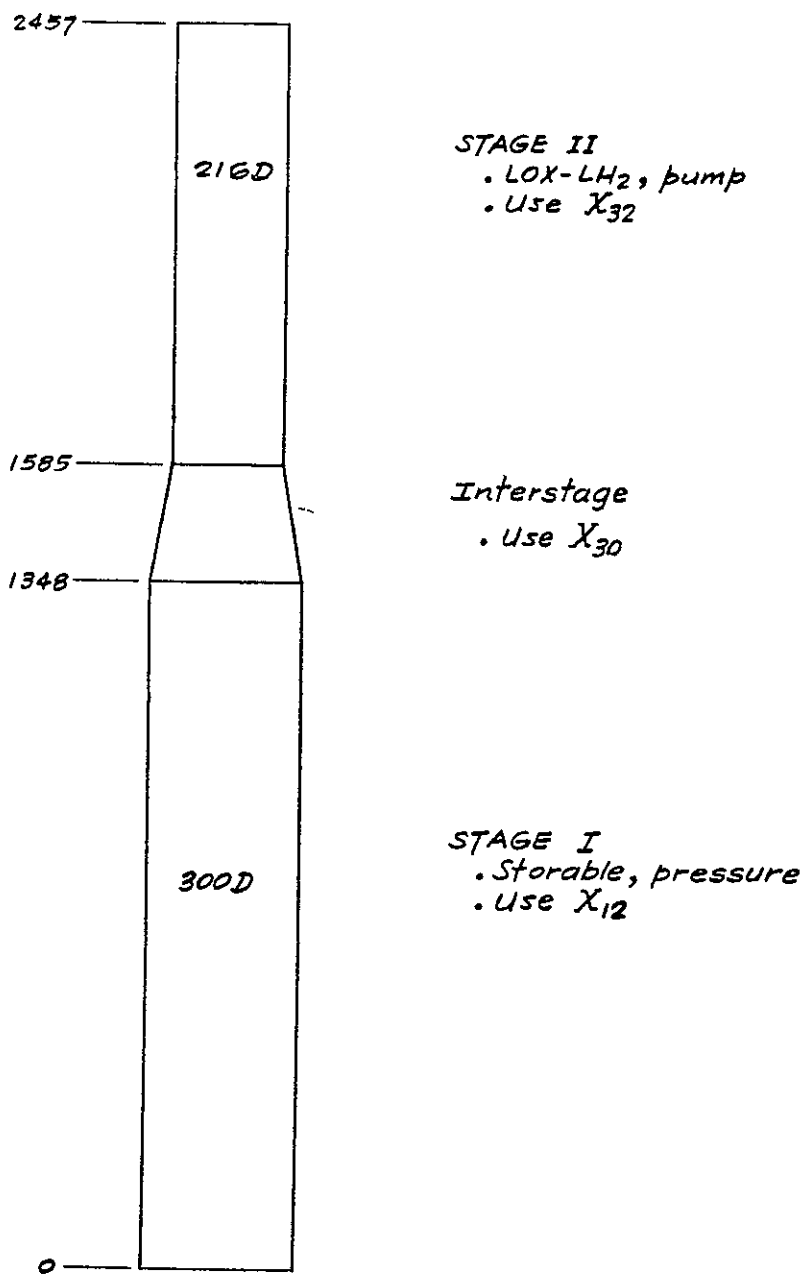
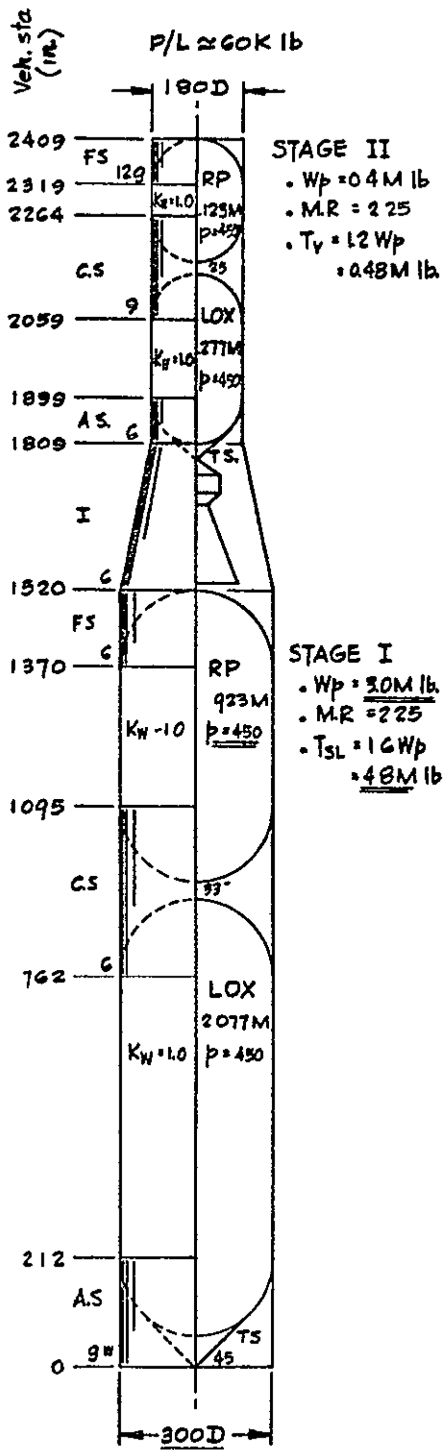


Figure 3-27

X_{17}. LOX-RP, PRESSURE



$N_c = \frac{14g(\Sigma Wt)}{\pi D}$

Veh. sta (in.)	ΣWt @BO	N_c(ULT) lb/in
2409	60K	
2319	64K	1905
2264	189K	
2059	199K	
1899	482K	4430
1809	487K	7250
1520	500K	4010
1370	530K	4250
1095	584K	4680
762	650K	5210
212	(758K)	-
0	$(800K)^{9W}$	6860

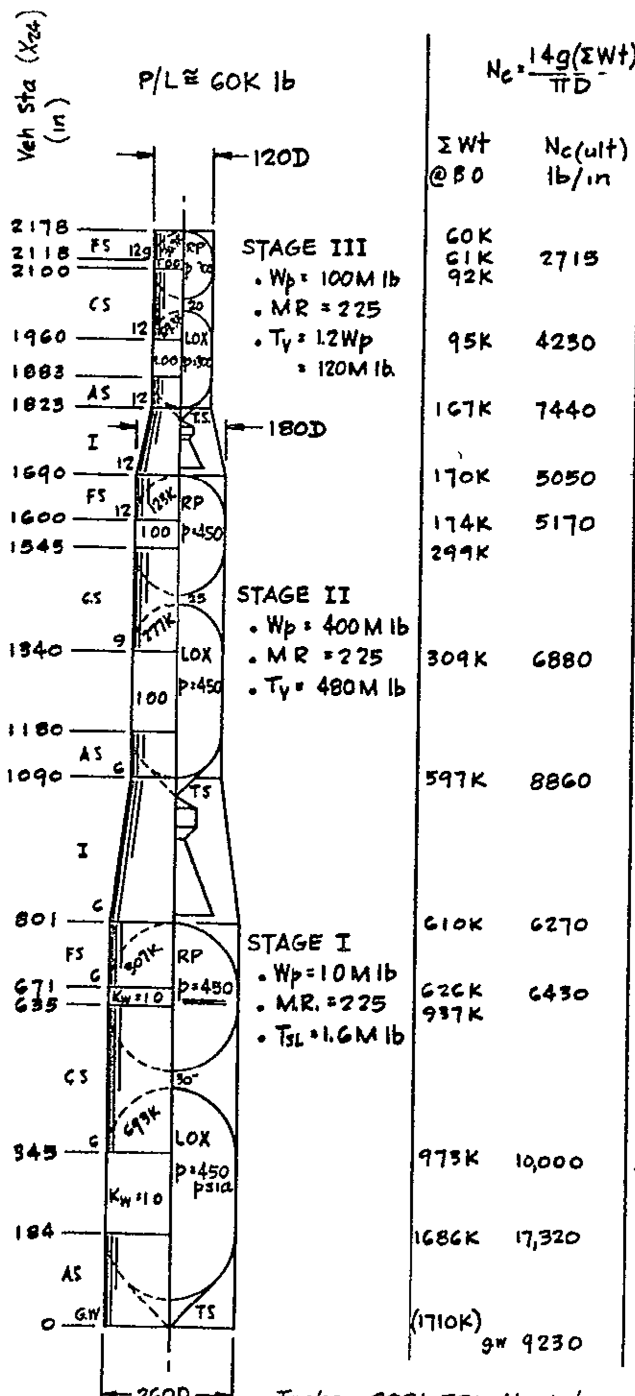
		Wt (lb)*	
FUEL TANK		7065	X_{18}
FWD DOME	2185		
CYL	2670		
AFT DOME	2210		
OXID TANK		12,175	X_{18}
FWD DOME	2185		
CYL	7750		
AFT DOME	2240		
FWD SKIRT		675	X_{27}
CTR SKIRT		2520	X_{27}
AFT SKIRT		1442	X_{27}
THRUST STR		1330	X_{27}
INTERSTAGE		5300	X_{27}
FUEL TANK		57,450	
FWD DOME	10,100		
CYL	36,950		
AFT DOME	10,400		
OXID TANK		94,740	
FWD DOME	10,100		
CYL	73,900		
AFT DOME	10,740		
FWD SKIRT		2910	X_{27}
CTR SKIRT		7275	X_{27}
AFT SKIRT		8220	X_{27}
THRUST STR		12,900	X_{27}

Tanks 2021-T81 Al plate, F_{tu} = 66 ksi, monocoque hemi-sph domes, monocoque sidewalls.
Skirts 7075-T6 plate, integ. mach. skin-stringer, internal frames.
Thrust Structures & Interstages. 7075-T6 alum. sht. & extr., conventional skin-stringer, internal frames.

* Non-optimum weight factors not included:
1.20 for interstages &
1.44 for all other structures.

Figure 3-28

X_{18}: LOX-RP, PRESSURE



		Wt(lb)*	
FUEL TANK		1126	
FWD DOME	432		
CYL	259		
AFT DOME	435		
OXID TANK		1980	
FWD DOME	432		
CYL	1107		
AFT DOME	441		
FWD SKIRT		361	X_{28}
CTR SKIRT		1080	X_{28}
AFT SKIRT		652	X_{28}
THR STRUCT		310	X_{28}
INTERSTAGE		1620	X_{28}
FUEL TANK		7065	
FWD DOME	2185		
CYL	2670		
AFT DOME	2210		
OXID TANK		12,175	
FWD DOME	2185		
CYL	7750		
AFT DOME	2240		
FWD SKIRT		1172	X_{28}
CTR SKIRT		3180	X_{28}
AFT SKIRT		1637	X_{28}
THR STRUCT		1480	X_{28}
INTERSTAGE		5825	X_{28}
FUEL TANK		16,903	
FWD DOME	6580		
CYL	3648		
AFT DOME	6675		
OXID TANK		29,660	
FWD DOME	6580		
CYL	16,300		
AFT DOME	6780		
FWD SKIRT		2795	X_{28}
CTR SKIRT		8220	X_{28}
AFT SKIRT		7440	X_{28}
THR STRUCT		4900	X_{28}

Tanks 2021-T81 Al plate, F_{tu} - 66 ksi, hemi-sph monocoque domes; monocoque sidewall

Skirts, thrust structures & interstages matls same as X_{27}

** Non-optimum wt. factors same as X_{27}*

Figure 3-29

X_{19} :

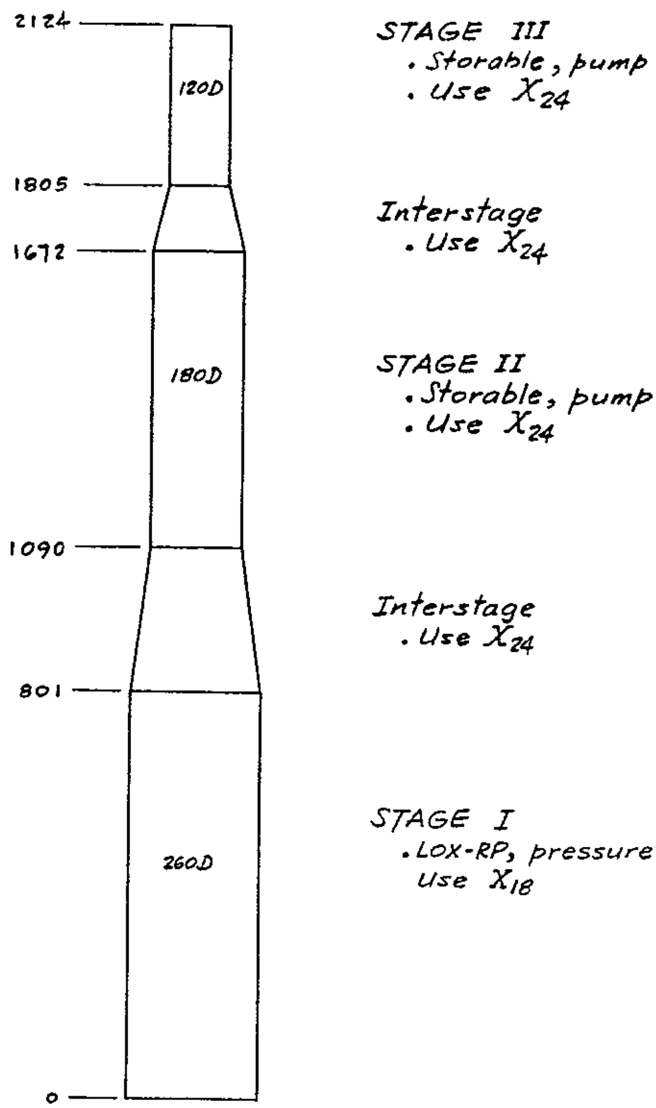


Figure 3-30

X_{20}:

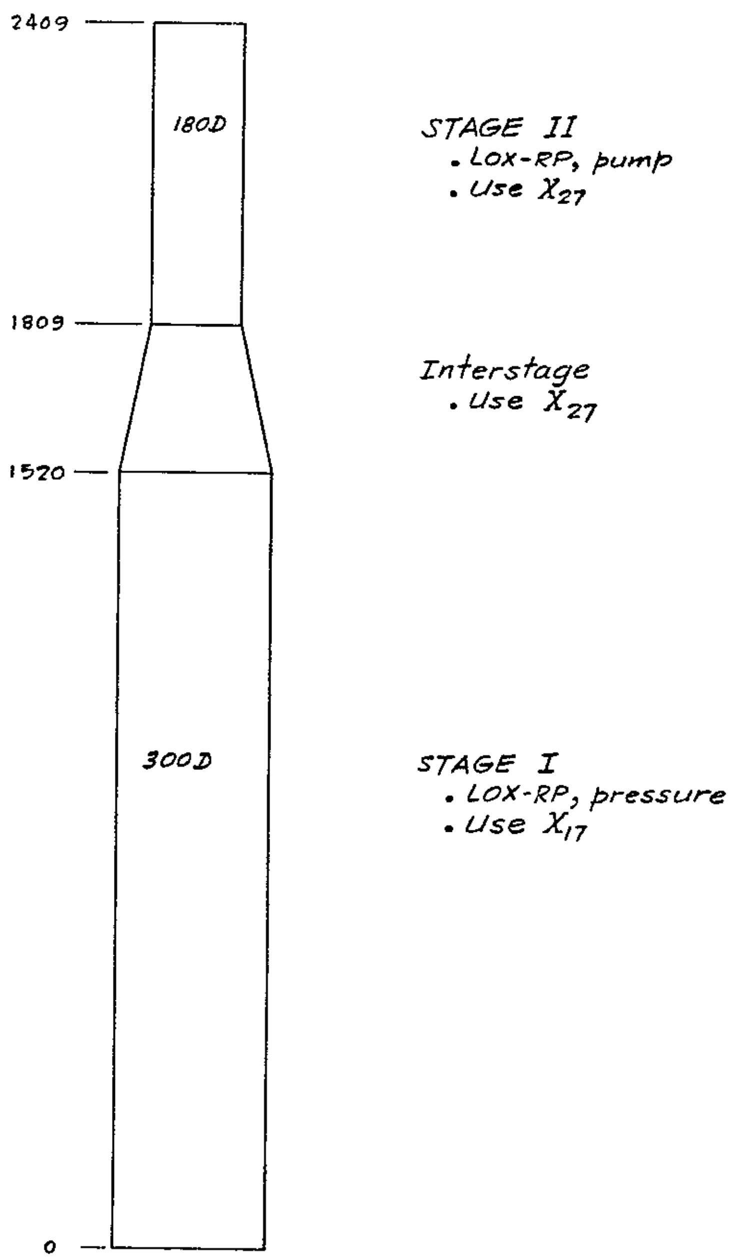


Figure 3-31

X_{21}:

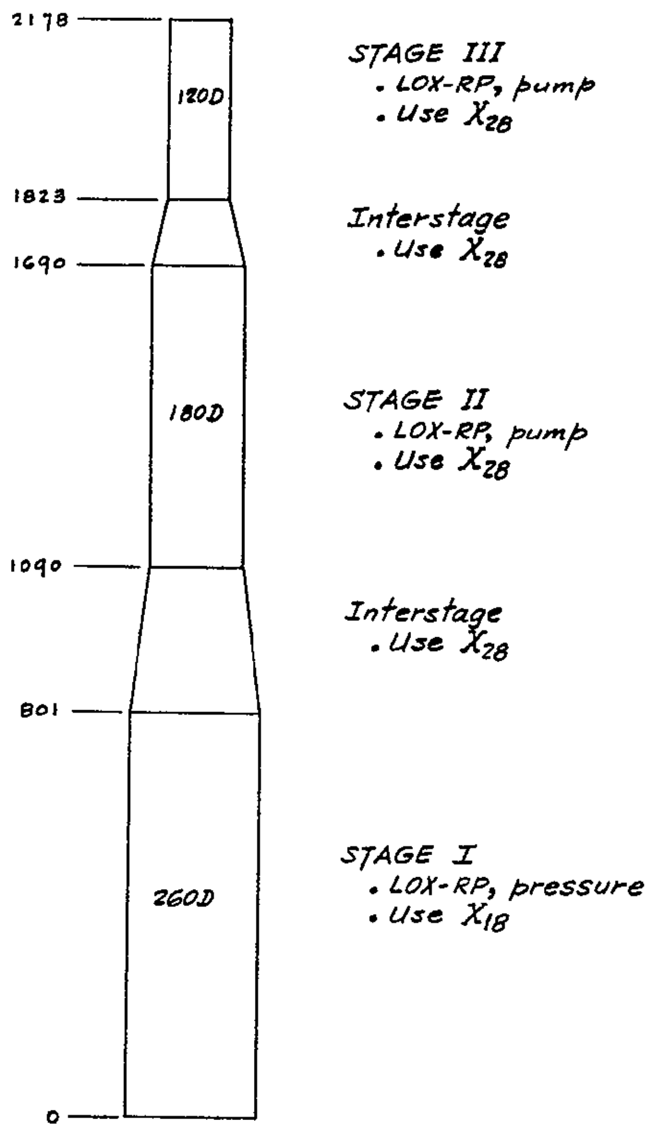


Figure 3-32

X_{22}:

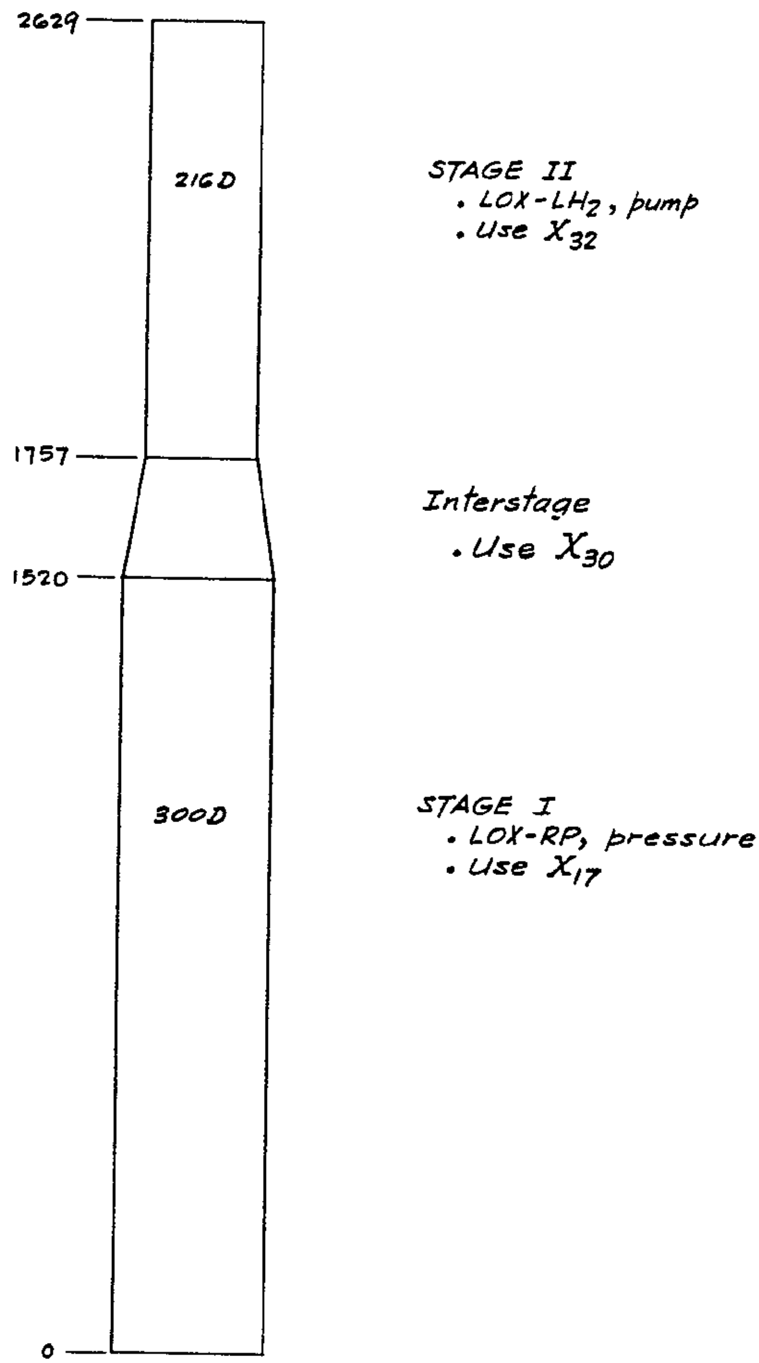
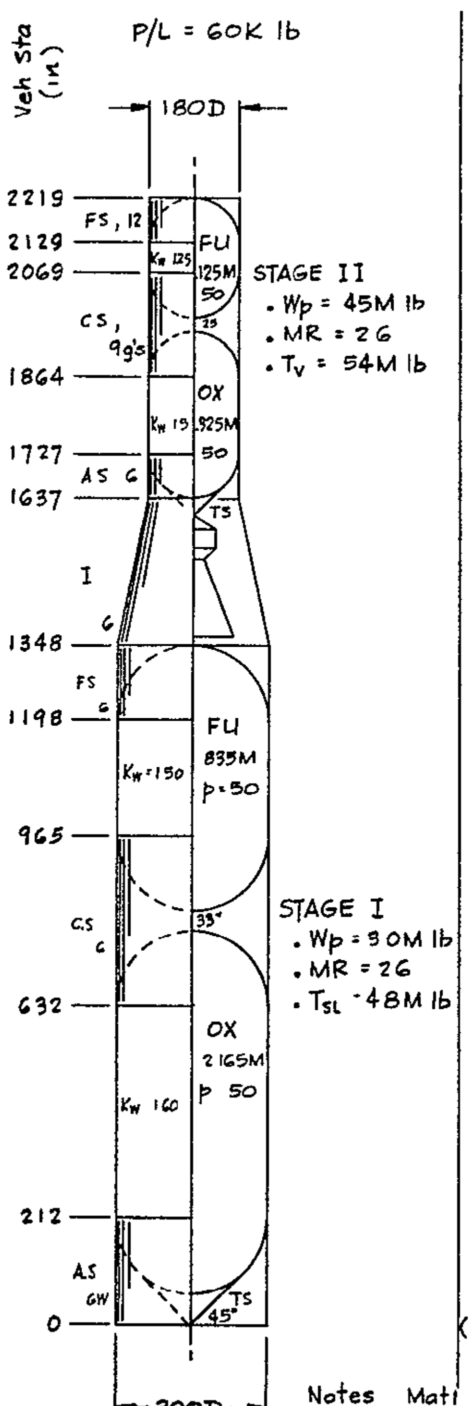


Figure 3-33

X_{23}: STORABLE, PUMP



$N_c = \frac{1.4g(\Sigma Wt)}{\pi D}$

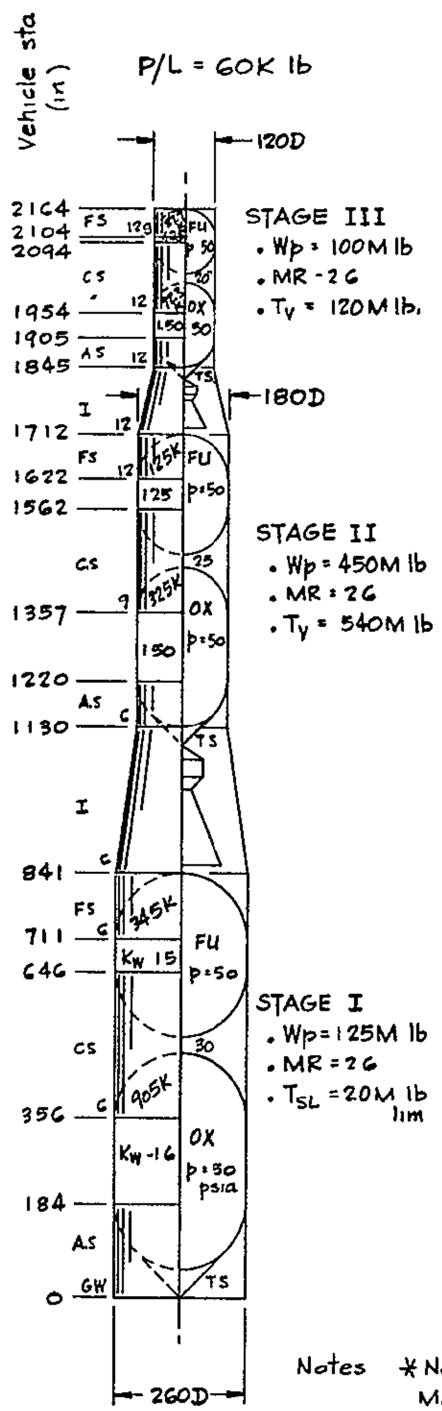
Veh sta (in)	Σ Wt @BO	N_c(ult) lb/in
2219	60K	
2129	65K	1930
2069	193K	
1864	203K	4520
1727	536K	
1637	541K	8030
1348	555K	4950
1198	588K	5240
965	646K	5700
632	714K	6370
212	807K	7190
0	(855K) Gr wind →	6860

		Wt* (lb)
FUEL TANK		916
FWD DOME	243	
CYL	405	
AFT DOME	268	
OXID TANK		1663
FWD DOME	243	
CYL	1110	
AFT DOME	310	
FWD SKIRT		678
CEN SKIRT		2485
AFT SKIRT		1537
THR STRUCT		1480
INTERSTAGE		5775
FUEL TANK		7736
FWD DOME	1124	
CYL	5230	
AFT DOME	1382	
OXID TANK		12,934
FWD DOME	1124	
CYL	10,060	
AFT DOME	1750	
FWD SKIRT		3290
CEN SKIRT		8190
AFT SKIRT		8220
THR STRUCT		12,900

Notes Matl & type of const same as X_{27}
* Non-optimum wt factors not included } 1.20 for interstage; 1.44 for all other components

Figure 3-34

X_{24}: STORABLE, PUMP



$N_c = \frac{14g(\Sigma Wt)}{\pi D}$

ΣWt @ BO	N_c(ult) lb/in
60K	
61K	2715
92K	4100
167K	7440
170K	5050
175K	5200
303K	
314K	6990
651K	9670
665K	6830
684K	7020
1039K	
1082K	11,120
2013K	-
(2040K)gw	9230

		Wt(lb)*	
FUEL TANK		179	
FWD DOME	72		X_{28}
CYL	30		
AFT DOME	77		X_{28}
OXID TANK		329	
FWD DOME	72		X_{28}
CYL	176		
AFT DOME	81		X_{28}
FWD SKIRT		361	X_{28}
CTR SKIRT		1060	
AFT SKIRT		652	X_{28}
THR STRUCT		310	X_{28}
INTERSTAGE		1620	X_{28}
FUEL TANK		916	X_{23}
FWD DOME	243		
CYL	405		
AFT DOME	268		
OXID TANK		1663	X_{23}
FWD DOME	243		
CYL	1110		
AFT DOME	310		
FWD SKIRT		1175	
CTR SKIRT		3220	
AFT SKIRT		1732	
THR STRUCT.		1480	X_{23}
INTERSTAGE		6125	
FUEL TANK		2654	
FWD DOME	732		X_{28}
CYL	1097		
AFT DOME	825		X_{28}
OXID TANK		4762	
FWD DOME	732		X_{28}
CYL	3100		
AFT DOME	930		X_{28}
FWD SKIRT		2950	
CTR SKIRT		8800	
AFT SKIRT		7440	X_{28}
THR STRUCT		5800	

Notes * Non-optimum factors same as X_{27}
Mat'l & type of const same as X_{27}

Figure 3-35

X_{25}:

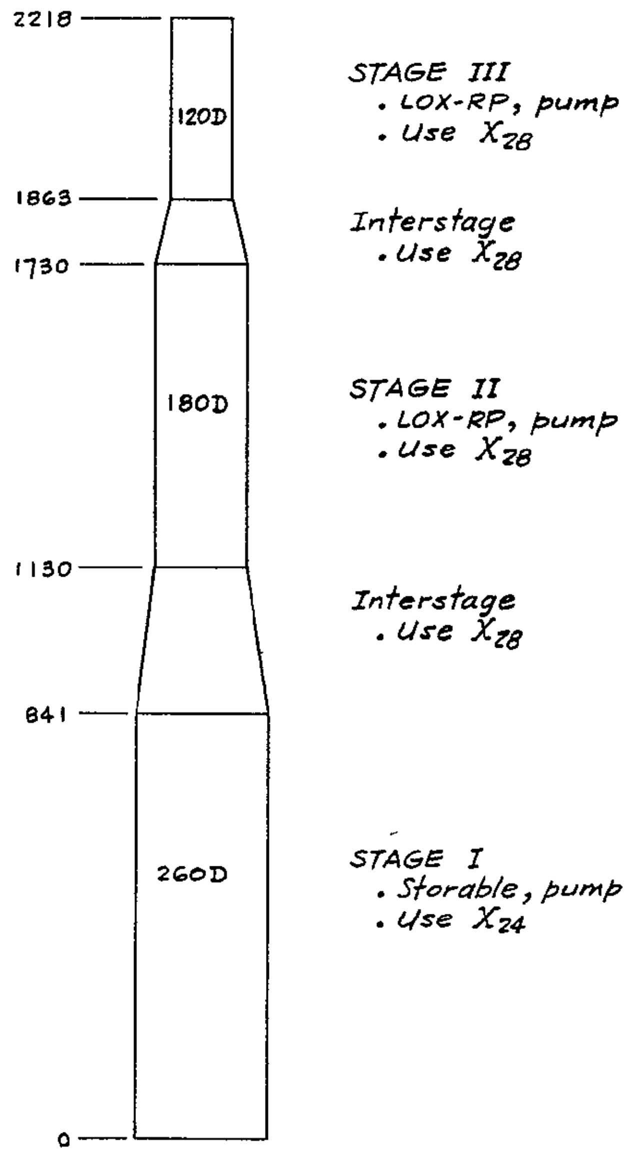


Figure 3-36

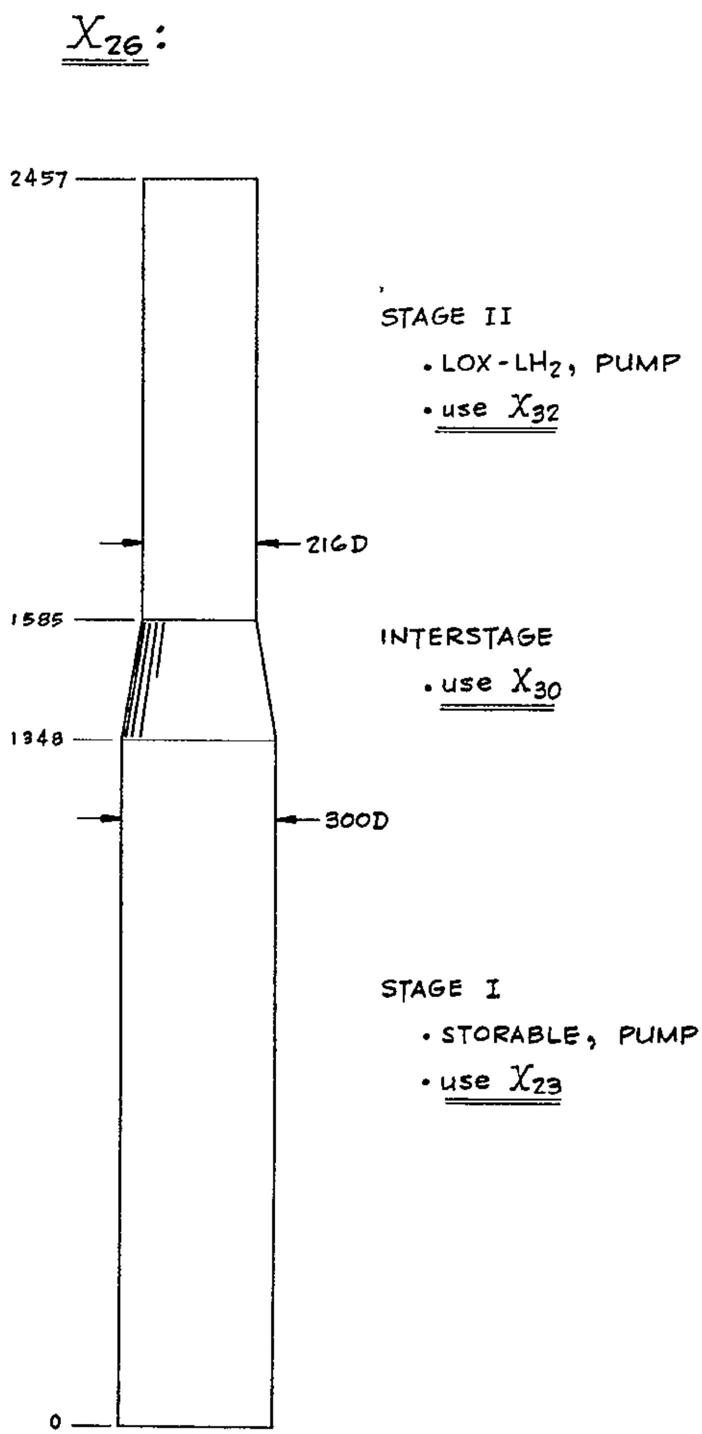
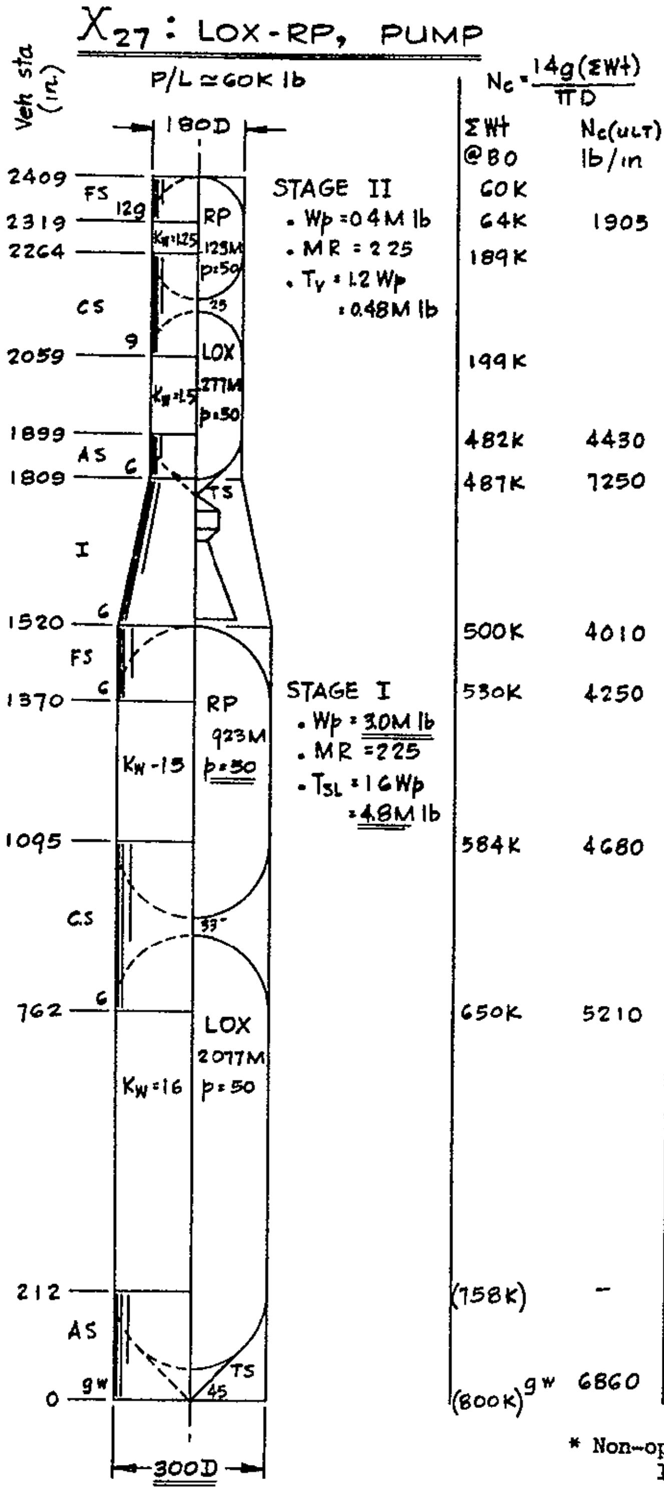

X_{26}:
2457
STAGE II
• LOX-LH_2, PUMP
• use X_{32}
216D
1585
INTERSTAGE
• use X_{30}
1348
300D
STAGE I
• STORABLE, PUMP
• use X_{23}
0

Figure 3-37



		Wt * (lb)	
FUEL TANK		882	X_{27}
FWD DOME	243		
CYL	371		
AFT DOME	268		
OXID TANK		1833	
FWD DOME	243		
CYL	1293		
AFT DOME	297		
FWD SKIRT		675	
CTR SKIRT		2520	
AFT SKIRT		1442	
THRUST STR		1330	
INTERSTAGE		5300	
FUEL TANK		8709	
FWD DOME	1124		X_{23}
CYL	6170		
AFT DOME	1415		
OXID TANK		16,014	
FWD DOME	1124		X_{23}
CYL	13,150		
AFT DOME	1740		
FWD SKIRT		2910	
CTR SKIRT		7275	
AFT SKIRT		8220	X_{25}
THRUST STR		12,900	X_{25}

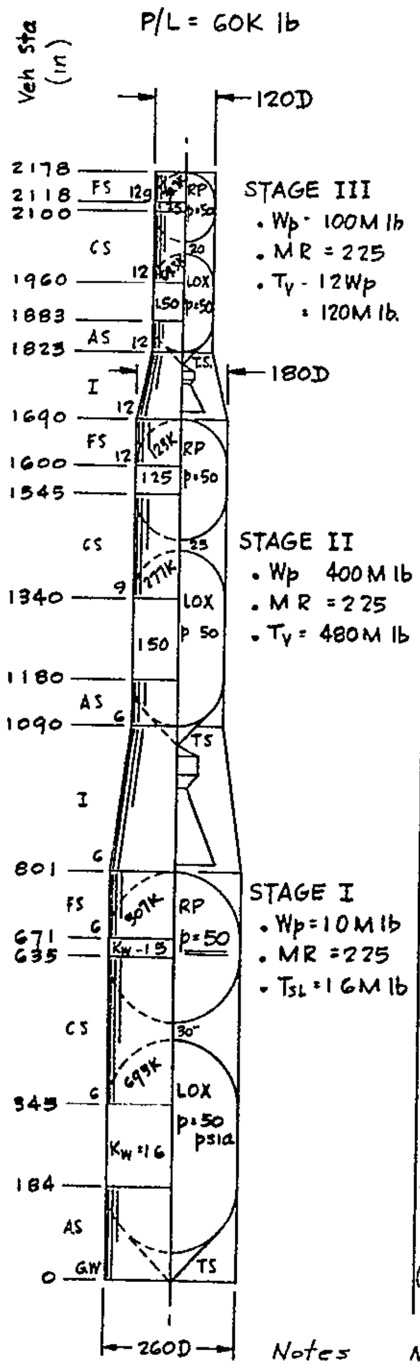
Tanks 2021-T81 plate, F_{tu}= 66ksi, hemi-spherical domes, waffle sidewall.
Skirts: 7075-T6 plate, integ. mach. skin-stringer, internal frames.
Thrust Structures & Interstages: 7075-T6 alum. sht & extr, conventional skin-stringer, internal frames.

* Non-optimum weight factors not included·
1.20 for interstages &
1.44 for all other structures.

Figure 3-38

X_{28}: LOX-RP, PUMP

P/L = 60K lb



$N_c = \frac{1.4g(\Sigma Wt)}{\pi D}$

ΣWt @BO	N_c(ult) lb/in
60K	
61K	2715
92K	
95K	4230
167K	7440
170K	5050
174K	5170
299K	
309K	6880
597K	8860
610K	6270
626K	6430
937K	
973K	10,000
1686K	17,320
(1710K) GW	9230

		Wt(lb)*	
FUEL TANK		203	
FWD DOME	72		
CYL	54		
AFT DOME	77		
OXID TANK		430	
FWD DOME	72		
CYL	277		
AFT DOME	81		
FWD SKIRT		361	
CTR SKIRT		1080	
AFT SKIRT		652	
THR STRUCT		310	
INTERSTAGE		1620	
FUEL TANK		882	X_{27}
FWD DOME	243		
CYL	371		
AFT DOME	268		
OXID TANK		1833	X_{27}
FWD DOME	243		
CYL	1293		
AFT DOME	297		
FWD SKIRT		1172	
CTR SKIRT		3180	
AFT SKIRT		1637	
THR STRUCT		1480	X_{27}
INTERSTAGE		5825	
FUEL TANK		2165	
FWD DOME	732		
CYL	608		
AFT DOME	825		
OXID TANK		4562	
FWD DOME	732		
CYL	2900		
AFT DOME	930		
FWD SKIRT		2795	
CTR SKIRT		8220	
AFT SKIRT		7440	
THR STRUCT		4900	

Notes Mat'l & type of const same as X_{27}
* Non-optimum factors same as X_{27}

Figure 3-39

$\underline{\underline{X_{30}}}$:

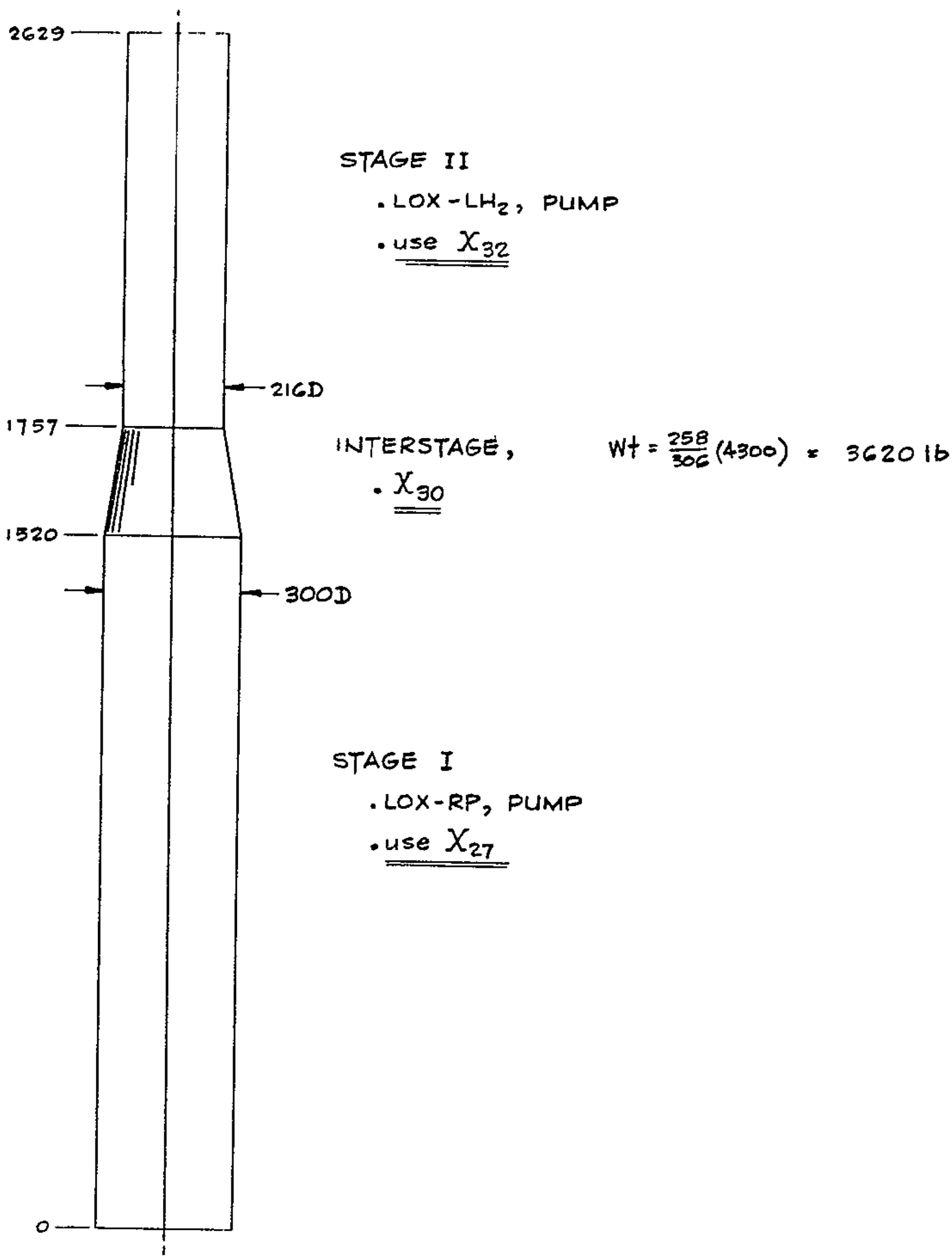
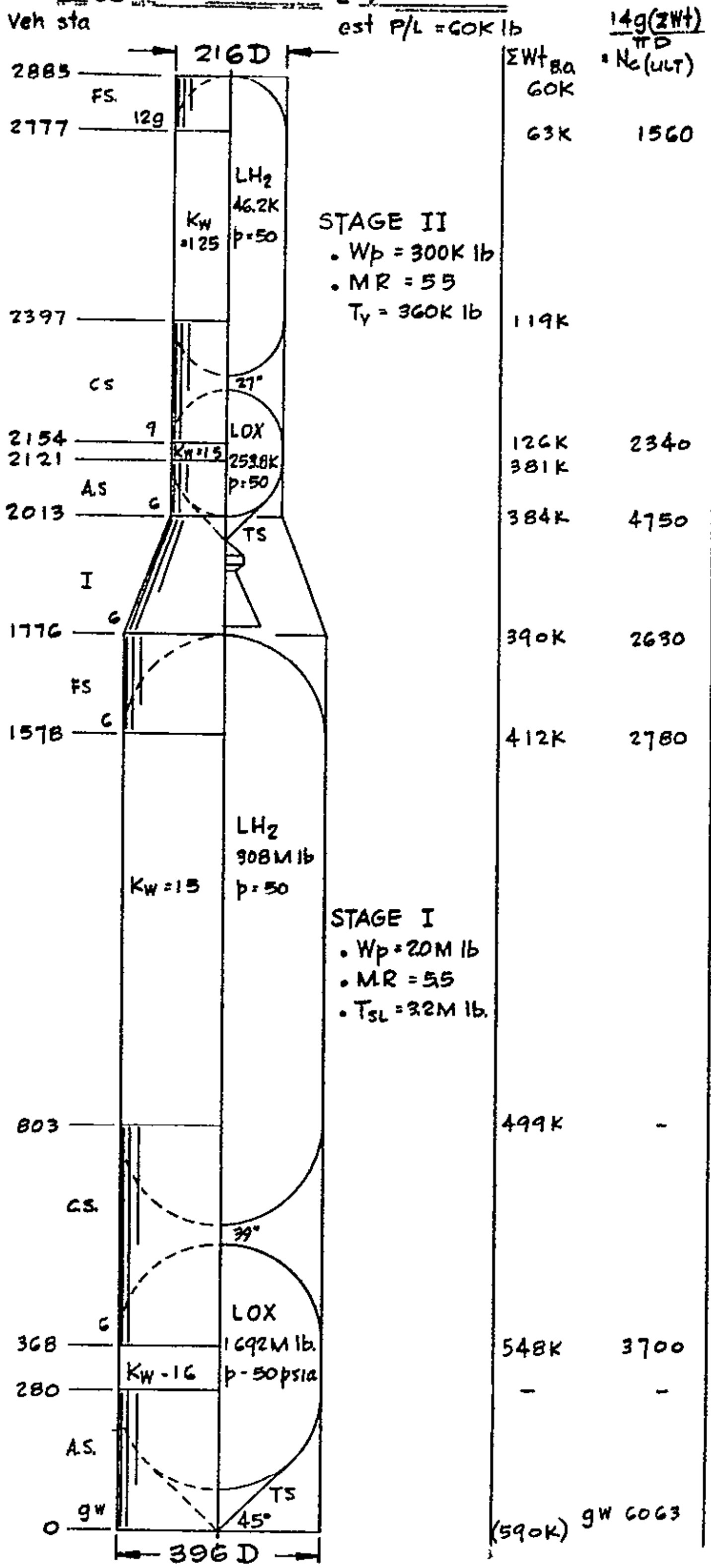


Figure 3-40

X_{32}: LOX-LH_2, PUMP



		Wt (lb)*
FUEL TANK		4540
FWD DOME	420	
CYL	3690	
AFT DOME	430	
OXID TANK		1287
FWD DOME	420	
CYL	385	
AFT DOME	482	
FWD SKIRT		890
CTR SKIRT		2435
AFT SKIRT		1610
THR STRUCT		1310
INTERSTAGE		4300
FUEL TANK		35,630
FWD DOME	2580	
CYL	30,350	
AFT DOME	2700	
OXID TANK		9595
FWD DOME	2580	
CYL	3675	
AFT DOME	3340	
FWD SKIRT		3990
CTR SKIRT		10,250
AFT SKIRT		13,245
THR STRUCT		12,500

*Non-optimum wt factors not incl
1.20 for interstages
1.44 for all other struct

Materials & types of construction same as X_{27}.

Figure 3-41

IV PARAMETRIC WEIGHT AND COST DATA FOR SIZING CANDIDATES

Weights and costs of each stage of each concept as a function of stage size (propellant weight) were prepared as inputs to the minimum cost sizing analysis These data are presented in this section Weights (Section 4 1) and Cost (Section 4 2)

4 1 Parametric Weight Analysis - Far too often erroneous conclusions are drawn from studies such as this because, in the name of expediency, weight data from previous studies are used as a basis for comparing alternate booster concepts An absolute requisite to any comparative evaluation of booster alternatives is the removal of all bias This dictates that all candidates be analyzed under a consistent set of ground rules and assumptions In the comparative evaluation of boosters, this consistency must be reflected in the mission, mission objectives, design loads and criteria, and the philosophy on which the design in predicated--in this case, low cost This emphasis on consistency, however, should not eclipse the importance of accuracy Estimates of system weight must be reasonably representative of real hardware or they could indirectly bias the evaluation While weight data from previous studies usually satisfies this latter requirement, two independent studies seldom if ever possesses the mutal consistency required for comparative evaluation Because of this, the approach taken here has been to rederive all weight data, using previous studies merely as a guide to design philosophy and as a basis for establishing appropriate design parameters, (e g tank pressures, engine parameters, etc) This approach, while possibly sacrificing something in the accuracy of total system weight, due to the limited depth of analysis feasible, does provide an objective comparison of candidates

To compensate for the limited depth of analysis that is feasible, abundant use has been made of previously derived empirical relationships These relationships, which depict subsystem weight as a function of appropriate design parameters, have been substantiated with hardware data and in most cases show reasonable correlation Another technique that was used to correlate estimates to actual hardware weight, was to apply so-called nonoptimum factors These are factors that are derived by comparing the weight estimate of an existing hardware system with its actual weight The resulting factor is used as a correction to the estimated weights of functionally similar proposed systems For subsystems where neither of these techniques appear appropriate or feasible, peg-point designs have been defined and detailed weight estimates made

Perturbations from these peg-points have been obtained by applying appropriate scaling laws

An example of the empirical representation of hardware weight is the relationship used to derive engine weights This equation has the form

$$W_{ENG} = K_1 \epsilon^{\alpha} \left[\frac{F}{C_f P_c}\right]^{\beta} + K_2 Q_v^{\gamma} \quad P_d^{\phi} \tag{4-1}$$

where

ϵ = nozzle area ratio

$\frac{F}{C_f P_c}$ = nozzle throat area

Q_v = total volumetric flowrate

P_d = mean pump discharge pressure

As can be noted, Equation 4-1 accounts for all major engine parameters, thrust, chamber pressure, area ratio, etc It has been derived and substantiated by correlation with the RL-10, BAC (8105), XLR-87, J-2, and F-1 engine weights This correlation is illustrated in Figure 4-1

An example of the application of nonoptimum factors and scaling law relationships is combined in the derivation of structure weight Here each propellant combination/stage concept has been analyzed in detail at a selected peg-point propellant load Selection of this peg-point propellant load is based on the nominal propellant load anticipated for the range of vehicles to be considered For example, for two-stage vehicles, the nominal LOX/RP-1 pump-fed first stage would require a propellant load of about 3,000,000 lb and the nominal LOX/RP-1 second stage a propellant load of about 400,000 lb Peg-point structure weights for these stages and all other candidates are presented in Section 3 3 Also included in that section is a discussion of the methodology used to derive the weights That analysis, however, considered only primary structure tanks, skirts, thrust structure, etc All secondary structure plus weight penalties for joints and discontinuity must be accounted for by applications of a nonoptimum factor The factor that has been applied to all calculated structure weights is 1 44 This factor was derived by comparing calculated weights of the Thor and S-IVB structure -- less joints, discontinuities, and

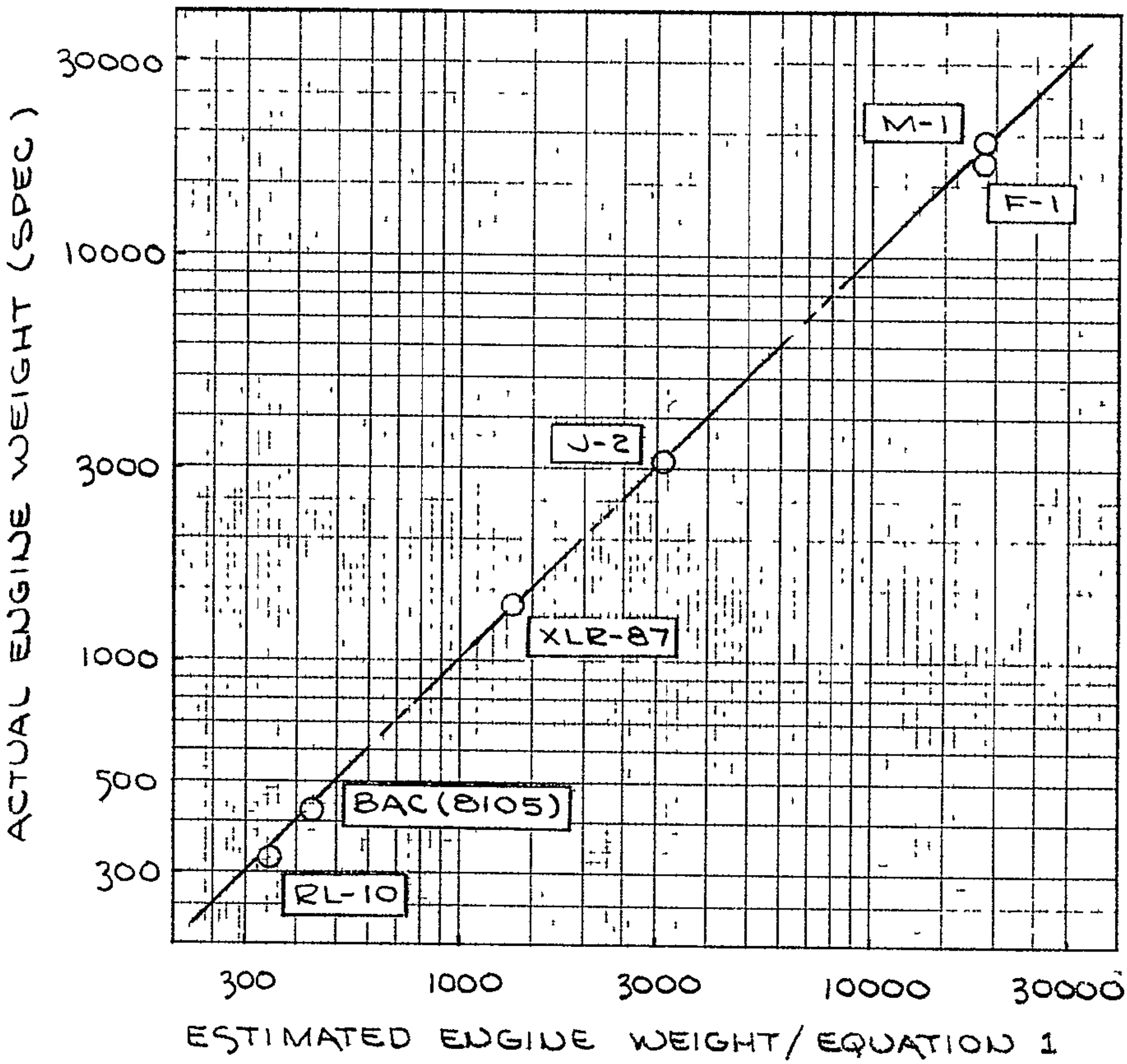


Figure 4-1 Correlation Engine Weight Equation

secondary structure -- with the actual weight of the total structure for each vehicle The factors obtained were 1 51 and 1 44, respectively The lower value obtained for S-IVB tends to indicate this factor may decrease with increased vehicle size However, considering the small sample size, (two vehicles) it was judged more prudent to select the more conservative assumption of a constant nonoptimum factor and use the value obtained from the larger more advanced vehicle

To perturb the weight of the peg-point structure over the range of propellant loads being considered, the following equation was used

$$W_{STR} = K_3 + K_4 (W_p) + K_5 (W_p)^{2/3} + K_6 (W_p)^{1/3} \tag{4-2}$$

In this equation, the first term accounts for those items of structure weight unaffected by tank volume and/or propellant weight The second term reflects those items of structure weight proportional to propellant load and/or tank volume, and the third and fourth terms that part of the structure weight proportional to surface area and the stage length respectively The weight details obtained from analysis of each peg-point design were used to derive appropriate values for K_3, K_4, K_5, and K_6 A unique set of coefficients was obtained for each propellant combination/stage concept considered

Output of this phase of the study consists of a plot of stage burnout weight versus propellant load and thrust, for each propellant/stage concept considered Since there are six propellant/engine concepts, including pump and pressure-fed options, and two- and three-stage booster concepts, 30 such plots were required Note that this figure is based on the assumption that the relative location of a stage, in the booster stack, significantly influences not only its design but also its weight As a result, parametric weights derived for the first stage of a two-stage candidate would not be applicable to either a second stage or the first stage of a three-stage candidate To facilitate this analysis, use has been made of an existing MDAC computer program This program, designated H756, was designed for the parametric analysis and preliminary sizing of liquid stages It has, built-in, all available company techniques for generating preliminary subsystem weight estimates In most cases these techniques are empirical (such as Equation 4-1) However, for subsystems inappropriate to empirical representation or where such representation is pending, scaling law equations similar to Equation 4-2 have been provided For subsystems where more than one estimating technique is available, the program provides the option most suitable to the data available Input to H756 consists of the propellant and thrust ranges to be considered, engine parameters desired, and scaling law constants for those systems not suited to empirical representation The output, of interest to this task, consists of a plot of step burnout weight and in-flight expendables plus a semi-detailed weight summary for each vehicle considered A sample of the tabular output is presented in Table 4-1 Summary plots for all candidates considered are shown in Figures 4-2 through 4-24 Each plot presents stage burnout weight, nonpropulsive expendables and aft interstage weight as a function of usable propellant Note that these plots reflect a fixed thrust to loaded propellant weight ratio This simplification has been adopted in an effort to eliminate the necessity of iterating thrust The assumption is that the thrust to propellant weight ratios indicated are commensurate with the desired thrust to gross vehicle weights If initial sizing substantiates this judgment no interation will be required

Table 4-1 Example Output (Program H756)

PRELIMINARY WEIGHT AND PERFORMANCE FOR PARAMETRIC SIZING STUDIES BD= 0 RR= 1 CASE= 5

MIXTURE RATIO-	2 27	ISP-	262.00
BULK DENSITY-	63.14	NUMBER OF ENGS.-	5

** WEIGHTS AND PERFORMANCE DATA FOR THRUST = .80000000+07 AND PROPELLANT WEIGHT = .50000000+07

ENG	FEED	FDVNT	PRESS	GMBL	RCS	CHLDN
[illegible]714+06	.15650310+05	.12929972+04	.11768192+05	.42077061+04	.13000000+04	.00000000
STR	PROPL	ASTR	PU	RESGS	RESVP	TRAP
12010666+06	.14232634+06	.82200000+03	.68849999+05	.00000000	.00000000	.27999999+05
START	VNTVP	VNTGS	RCP	MISC		
.00000000	.00000000	.00000000	.26000000+04	.00000000		
WBO	USABL	EXPND	GROSS	LMBDA		
[illegible]10500+06	.49031501+07	.26000000+04	.52718550+07	.93006164-00		

PL(1,3,5,...)	GSTG	VEL	PL(2,4,6,...)	GSTG	VEL
.00000000	.52718550+07	.22489787+05	.10000000+06	.53718550+07	.20611998+05
.20000000+06	.54718550+07	.19128618+05	.30000000+06	.55718550+07	.17909720+05
.40000001+06	.56718550+07	.16880319+05	.50000000+06	.57718550+07	.15993199+05
.60000000+06	.58718549+07	.15216708+05	.70000000+06	.59718550+07	.14528599+05
.80000000+06	.60718550+07	.13912640+05	.90000001+06	.61718550+07	.13356632+05
.10000000+07	.62718550+07	.12851174+05			

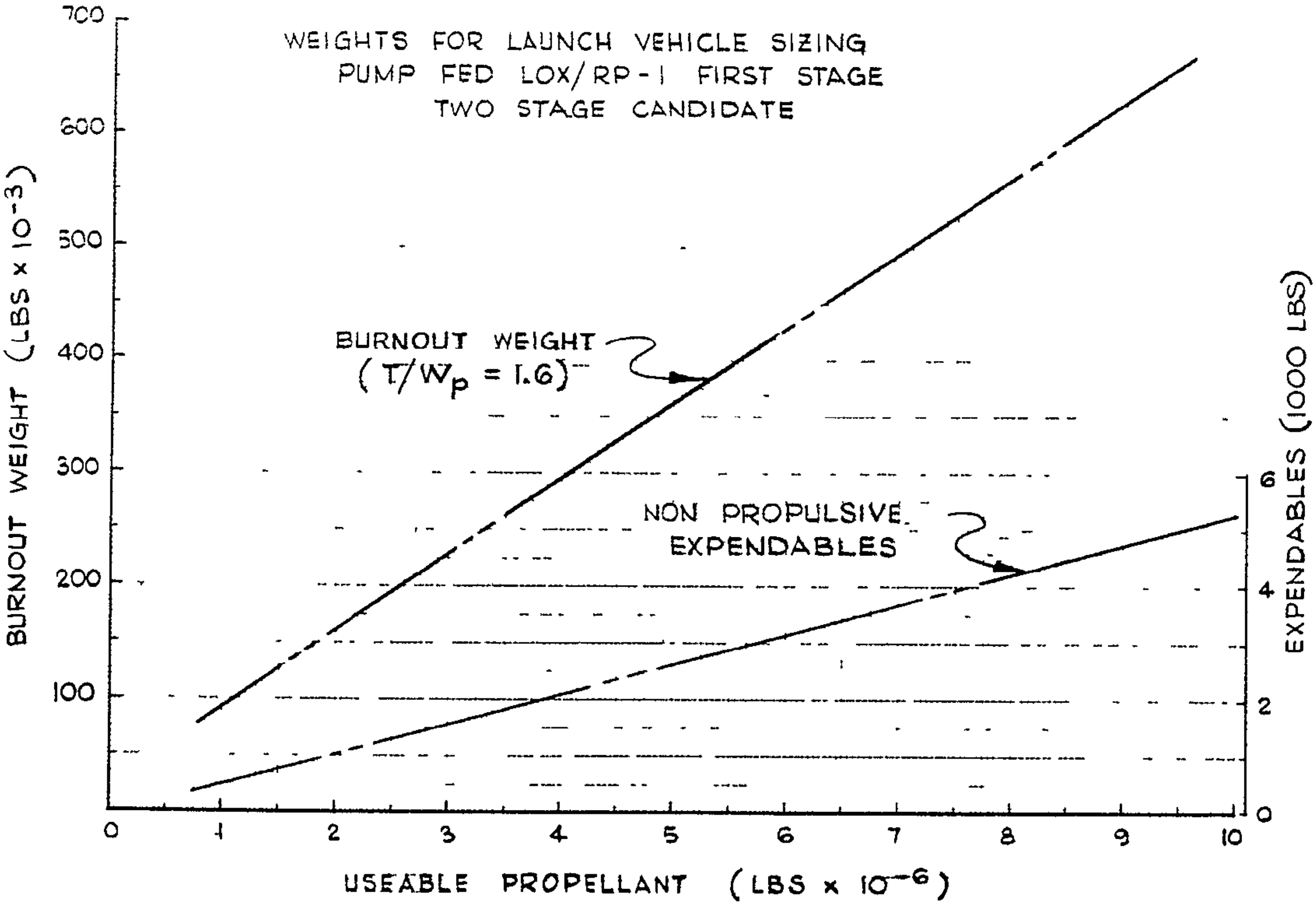

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LOX/RP-1 FIRST STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT
(T/Wp = 1.6)
NON PROPULSIVE
EXPENDABLES
BURNOUT WEIGHT (LBS x 10^-3)
EXPENDABLES (1000 LBS)
USEABLE PROPELLANT (LBS x 10^-6)
700
600
500
400
300
200
100
0
1
2
3
4
5
6
7
8
9
10

Figure 4-2

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LOX/RP-1 SECOND STAGE
TWO STAGE CANDIDATE

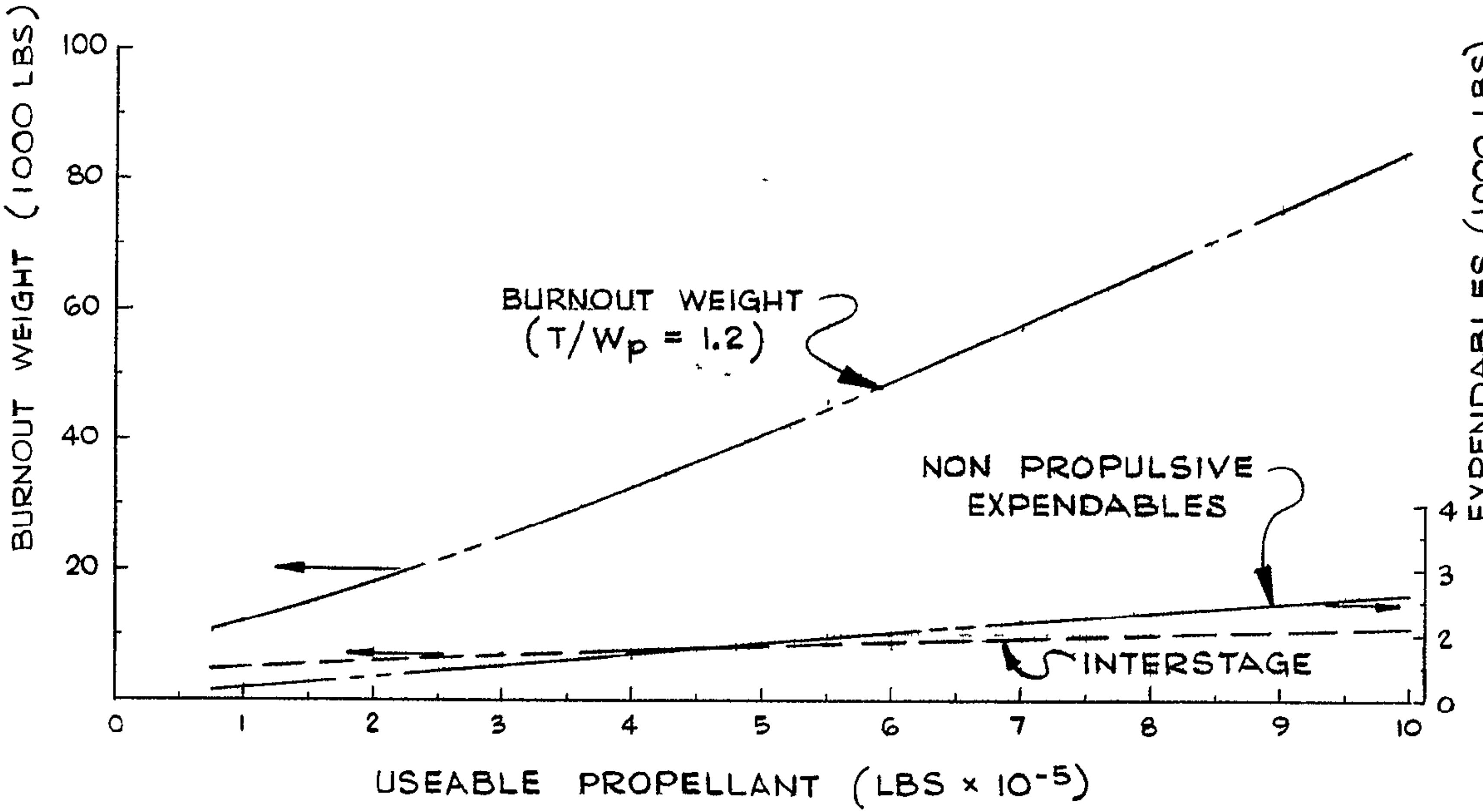


Figure 4-3

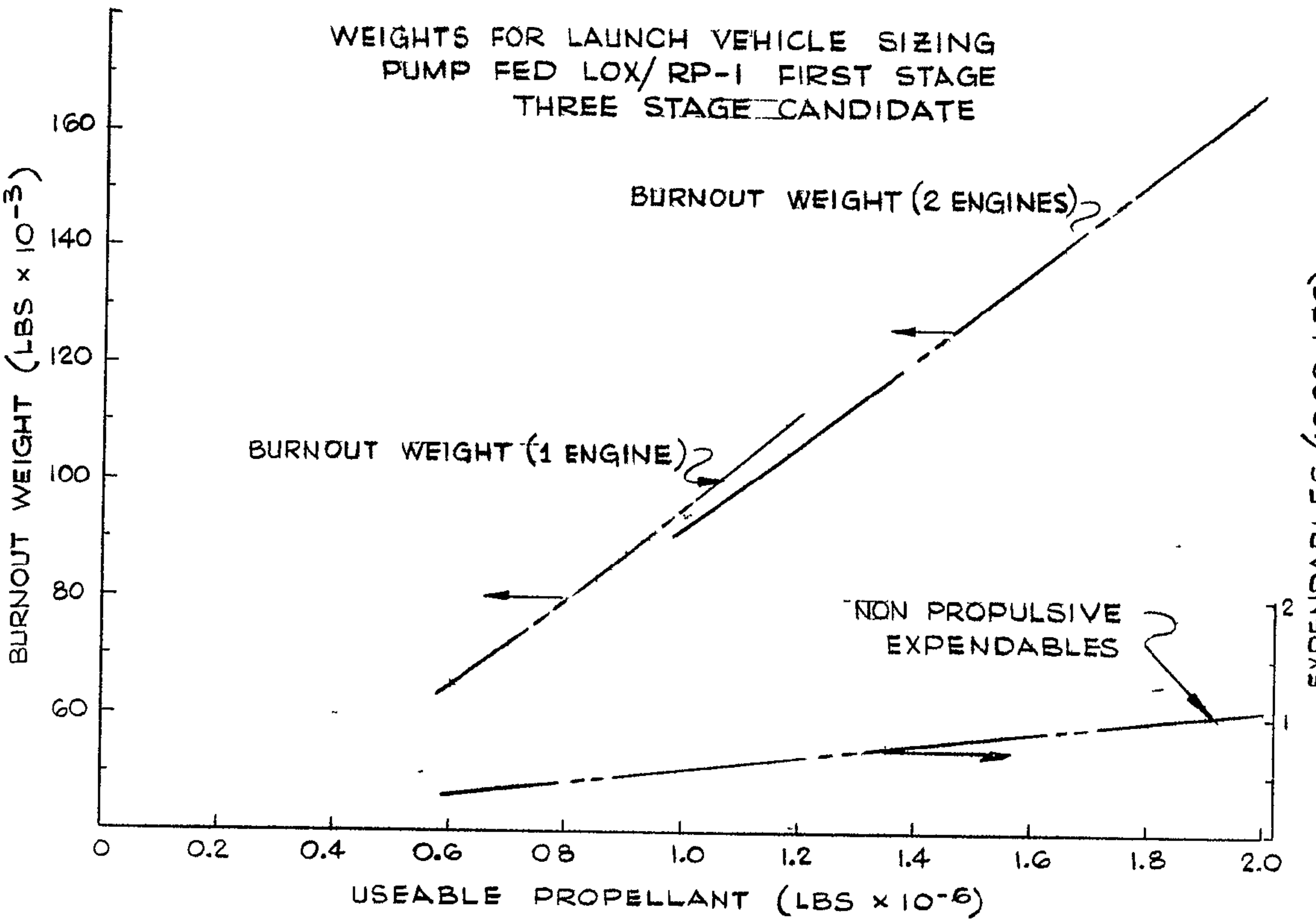

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LOX/ RP-1 FIRST STAGE
THREE STAGE CANDIDATE
BURNOUT WEIGHT (2 ENGINES)
BURNOUT WEIGHT (1 ENGINE)
NON PROPULSIVE EXPENDABLES
BURNOUT WEIGHT (LBS × 10⁻³)
160
140
120
100
80
60
EXPENDABLES (1000 LBS)
2
1
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
USEABLE PROPELLANT (LBS × 10⁻⁶)

Figure 4-4

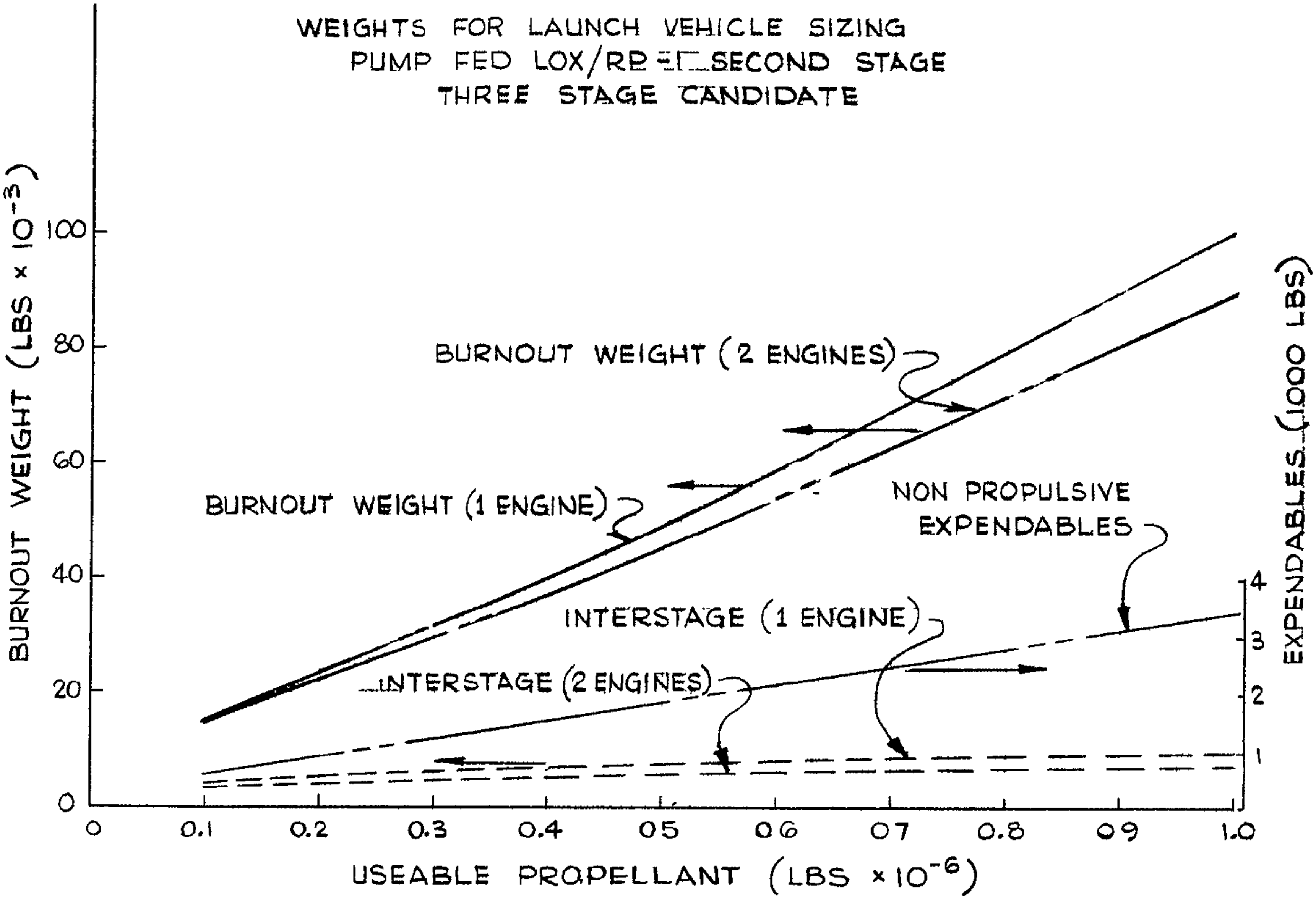

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LOX/RP-1 SECOND STAGE
THREE STAGE CANDIDATE
BURNOUT WEIGHT (2 ENGINES)
BURNOUT WEIGHT (1 ENGINE)
NON PROPULSIVE EXPENDABLES
INTERSTAGE (1 ENGINE)
INTERSTAGE (2 ENGINES)
BURNOUT WEIGHT (LBS × 10⁻³)
0
20
40
60
80
100
EXPENDABLES (1000 LBS)
1
2
3
4
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
USEABLE PROPELLANT (LBS × 10⁻⁶)

Figure 4-5

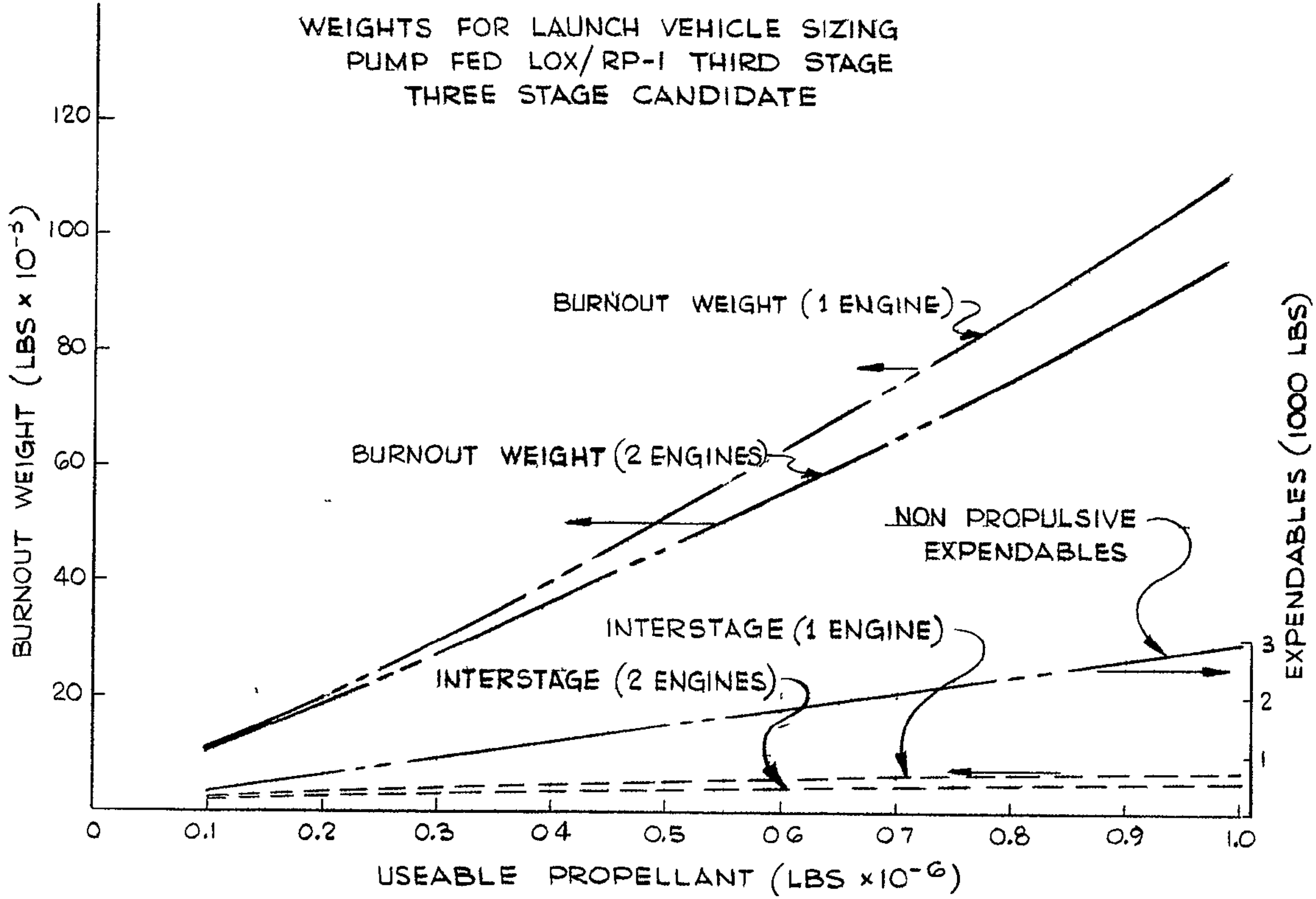

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LOX/RP-1 THIRD STAGE
THREE STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10⁻³)
120
100
80
60
40
20
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
USEABLE PROPELLANT (LBS x10⁻⁶)
EXPENDABLES (1000 LBS)
1
2
3
BURNOUT WEIGHT (1 ENGINE)
BURNOUT WEIGHT (2 ENGINES)
NON PROPULSIVE EXPENDABLES
INTERSTAGE (1 ENGINE)
INTERSTAGE (2 ENGINES)

Figure 4-6

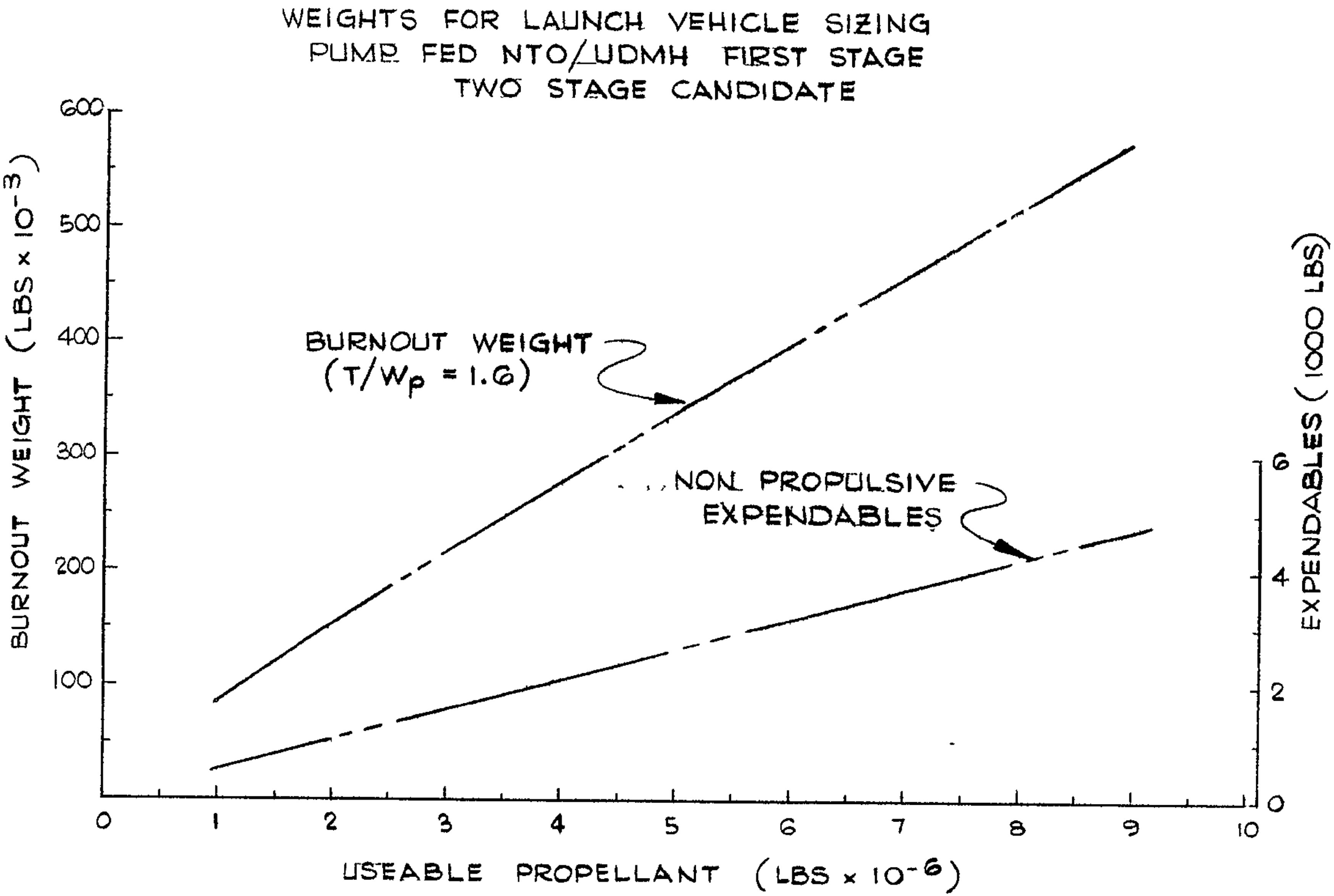

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED NTO/UDMH FIRST STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10⁻³)
EXPENDABLES (1000 LBS)
BURNOUT WEIGHT
(T/Wp = 1.6)
NON PROPULSIVE
EXPENDABLES
USEABLE PROPELLANT (LBS x 10⁻⁶)
600
500
400
300
200
100
0
1
2
3
4
5
6
7
8
9
10
2
4
6

Figure 4-7

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED NTO/UDMH SECOND STAGE
TWO STAGE CANDIDATE

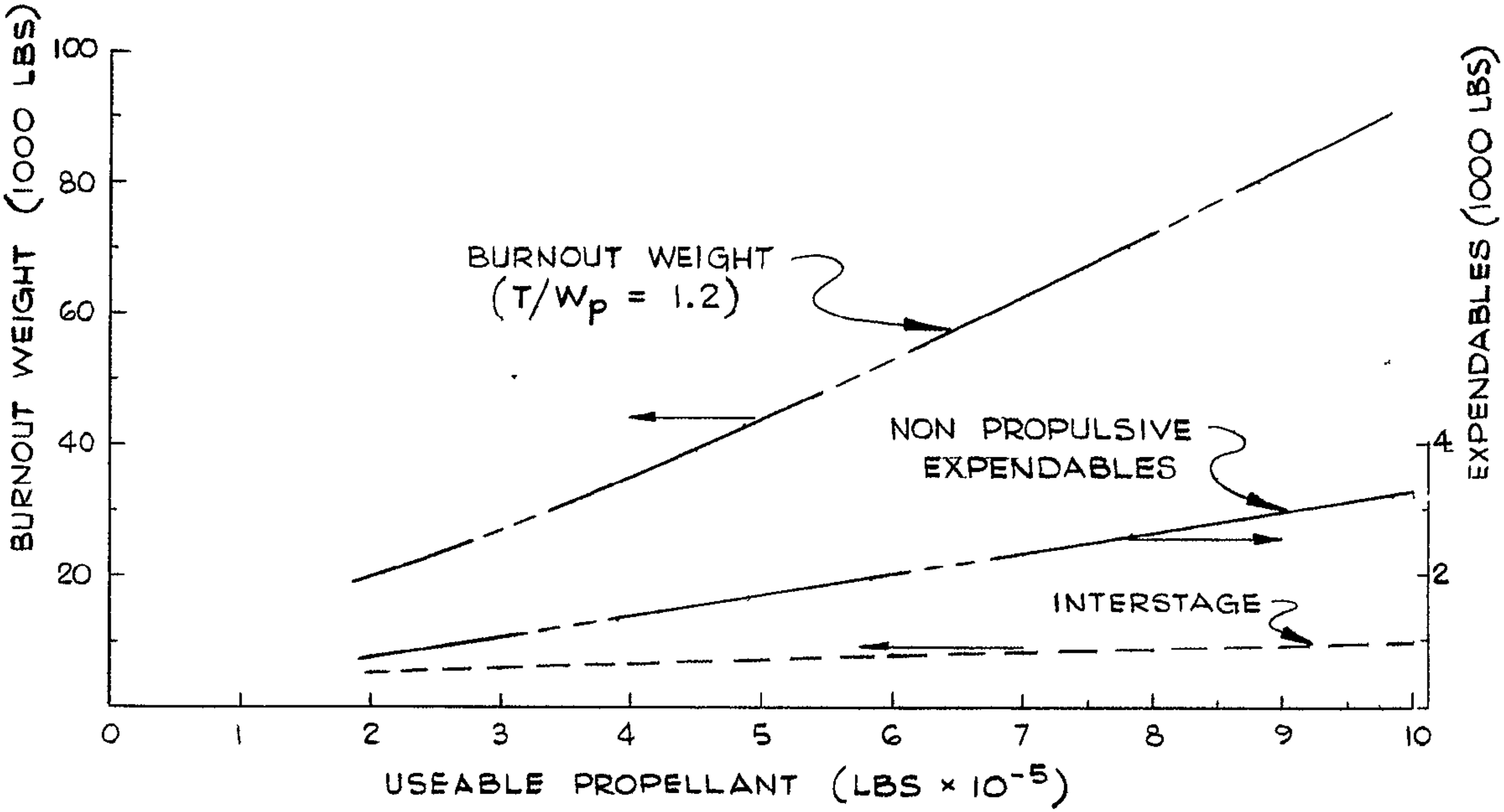


Figure 4-8

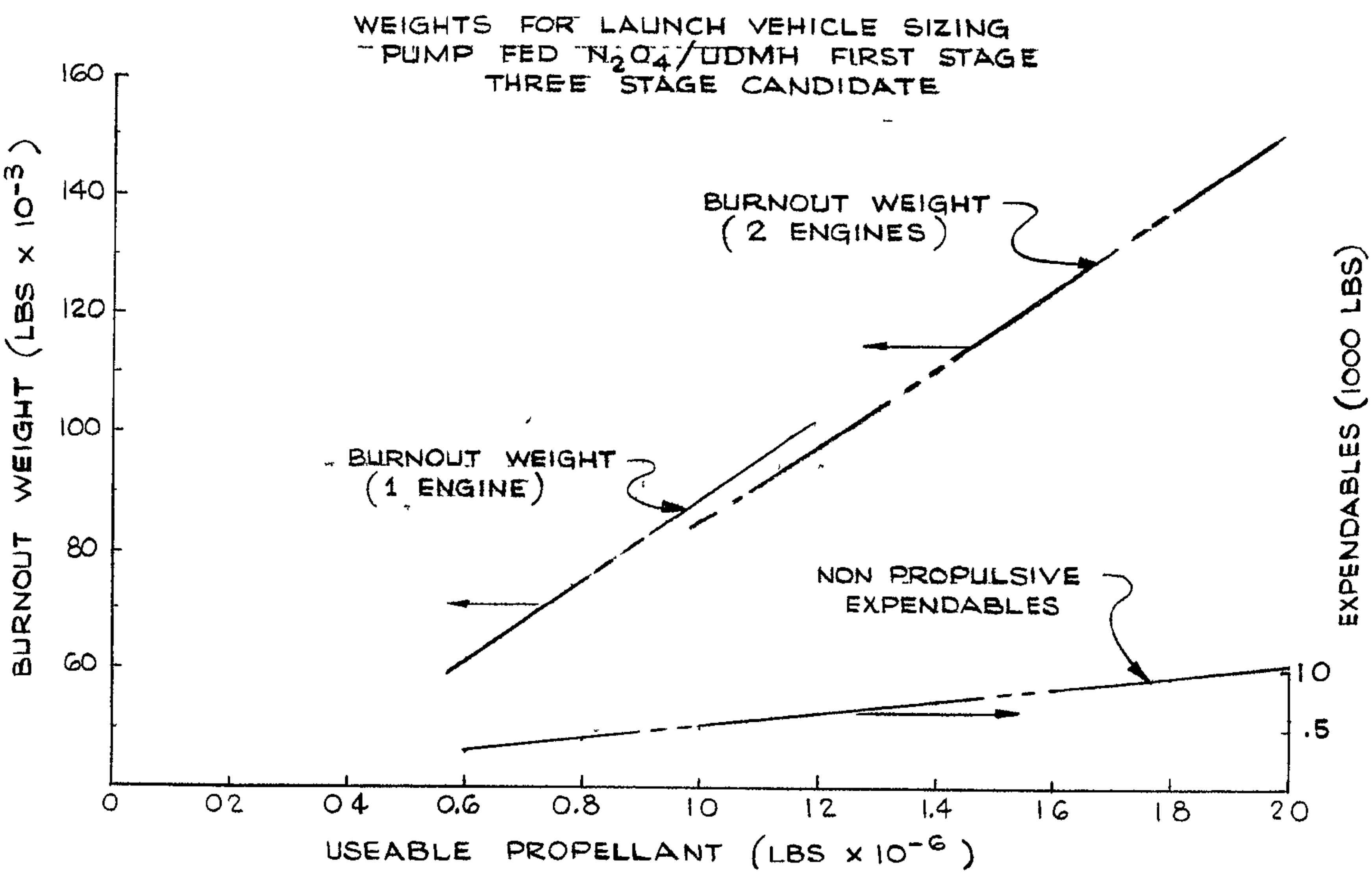

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED N_2O_4/UDMH FIRST STAGE
THREE STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10^{-3})
160
140
120
100
80
60
BURNOUT WEIGHT (2 ENGINES)
BURNOUT WEIGHT (1 ENGINE)
NON PROPULSIVE EXPENDABLES
EXPENDABLES (1000 LBS)
1.0
.5
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
USEABLE PROPELLANT (LBS x 10^{-6})

Figure 4-9

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED N_2O_4/UDMH SECOND STAGE
THREE STAGE CANDIDATE

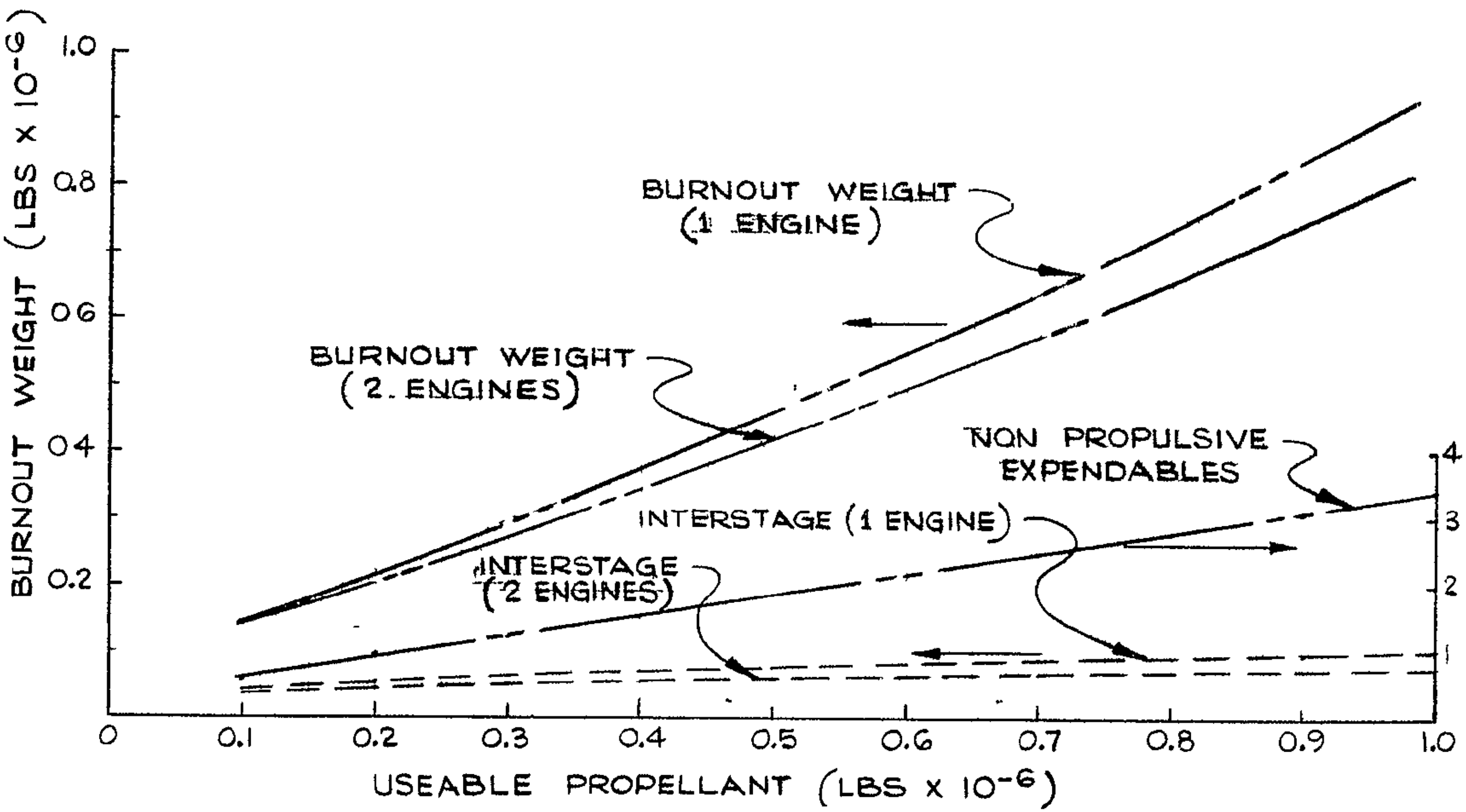


Figure 4-10

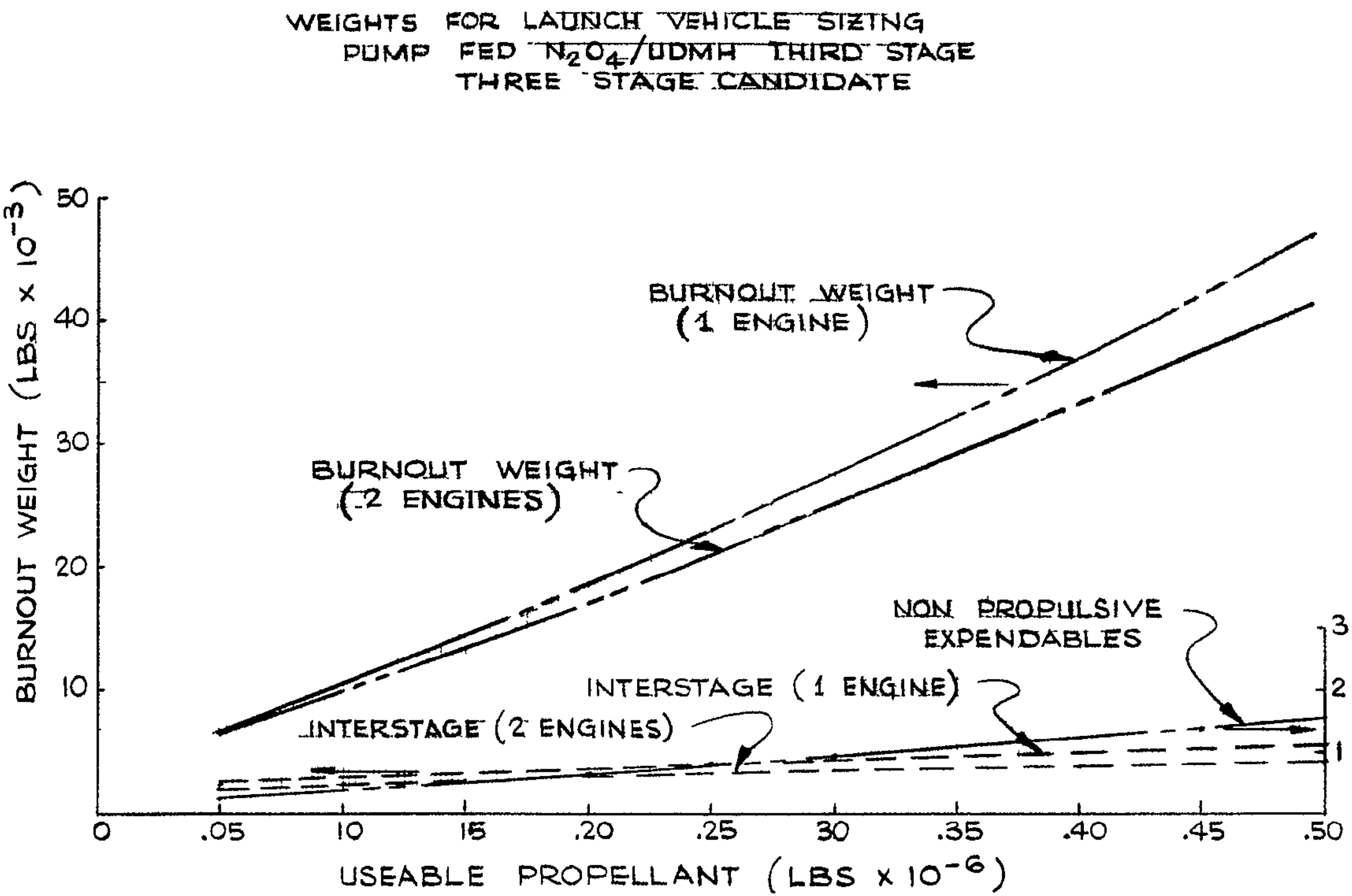

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED N_2O_4/UDMH THIRD STAGE
THREE STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10^{-3})
EXPENDABLES (1000 LBS)
BURNOUT WEIGHT (1 ENGINE)
BURNOUT WEIGHT (2 ENGINES)
NON PROPULSIVE EXPENDABLES
INTERSTAGE (1 ENGINE)
INTERSTAGE (2 ENGINES)
50
40
30
20
10
3
2
1
0
.05
.10
.15
.20
.25
.30
.35
.40
.45
.50
USEABLE PROPELLANT (LBS x 10^{-6})

Figure 4-11

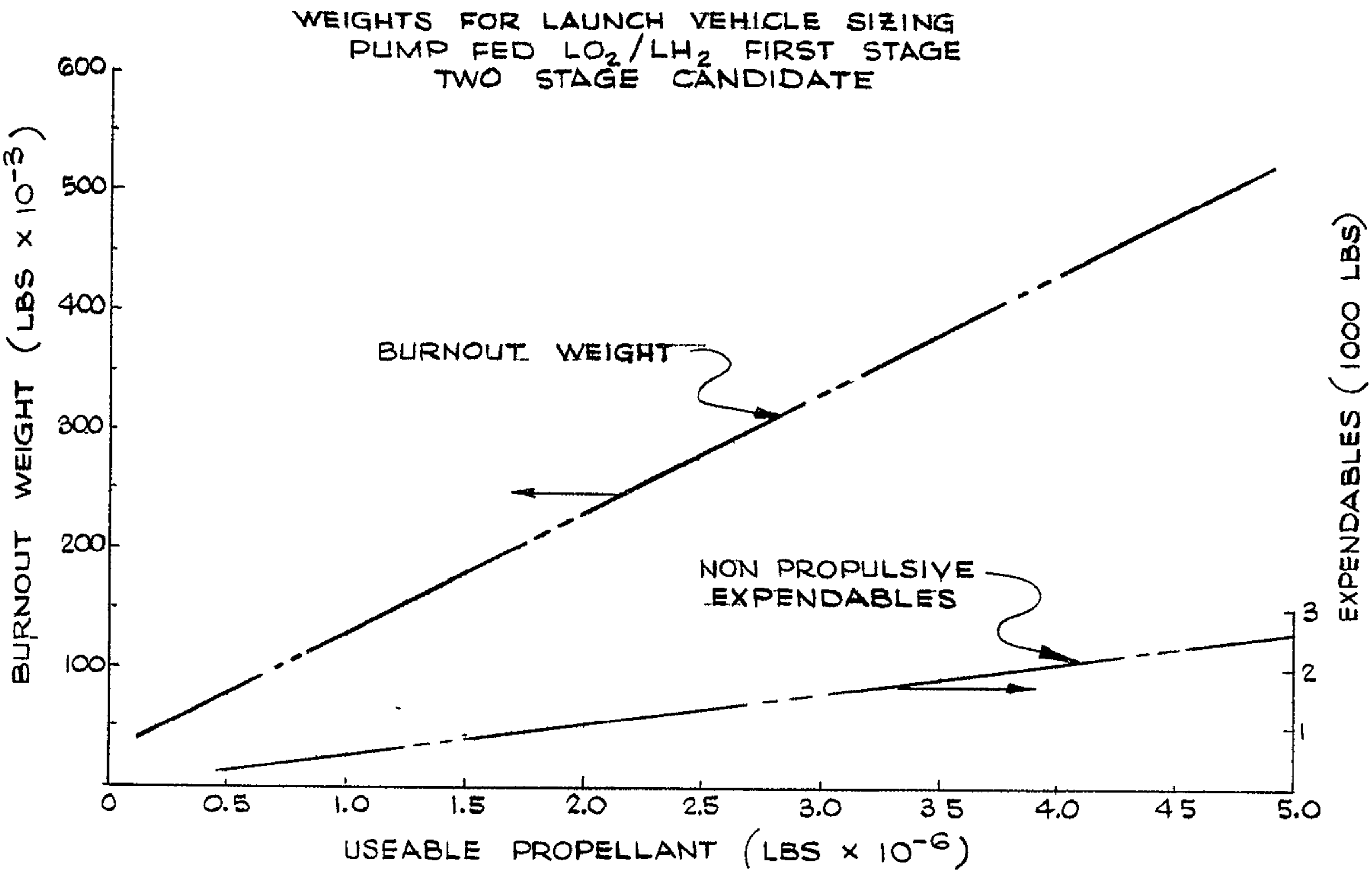

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LO_2/LH_2 FIRST STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT ($LBS \times 10^{-3}$)
600
500
400
300
200
100
BURNOUT WEIGHT
NON PROPULSIVE EXPENDABLES
EXPENDABLES (1000 LBS)
3
2
1
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
5.0
USEABLE PROPELLANT ($LBS \times 10^{-6}$)

Figure 4-12

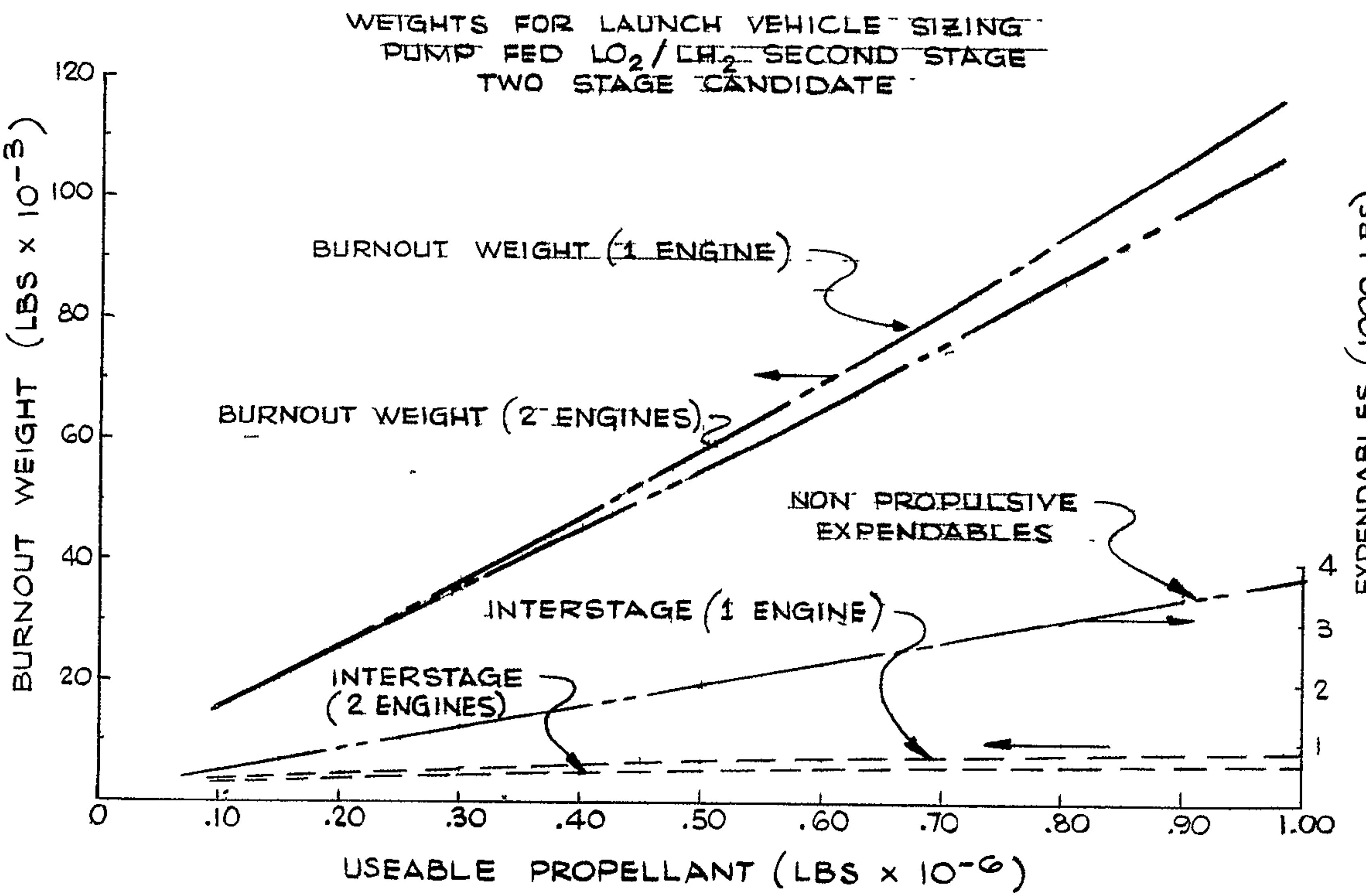

WEIGHTS FOR LAUNCH VEHICLE SIZING
PUMP FED LO_2/LH_2 SECOND STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10^{-3})
120
100
80
60
40
20
BURNOUT WEIGHT (1 ENGINE)
BURNOUT WEIGHT (2 ENGINES)
NON PROPULSIVE EXPENDABLES
INTERSTAGE (1 ENGINE)
INTERSTAGE (2 ENGINES)
EXPENDABLES (1000 LBS)
4
3
2
1
0
.10
.20
.30
.40
.50
.60
.70
.80
.90
1.00
USEABLE PROPELLANT (LBS x 10^{-6})

Figure 4-13

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED LOX/RP-1 FIRST STAGE
TWO STAGE CANDIDATE

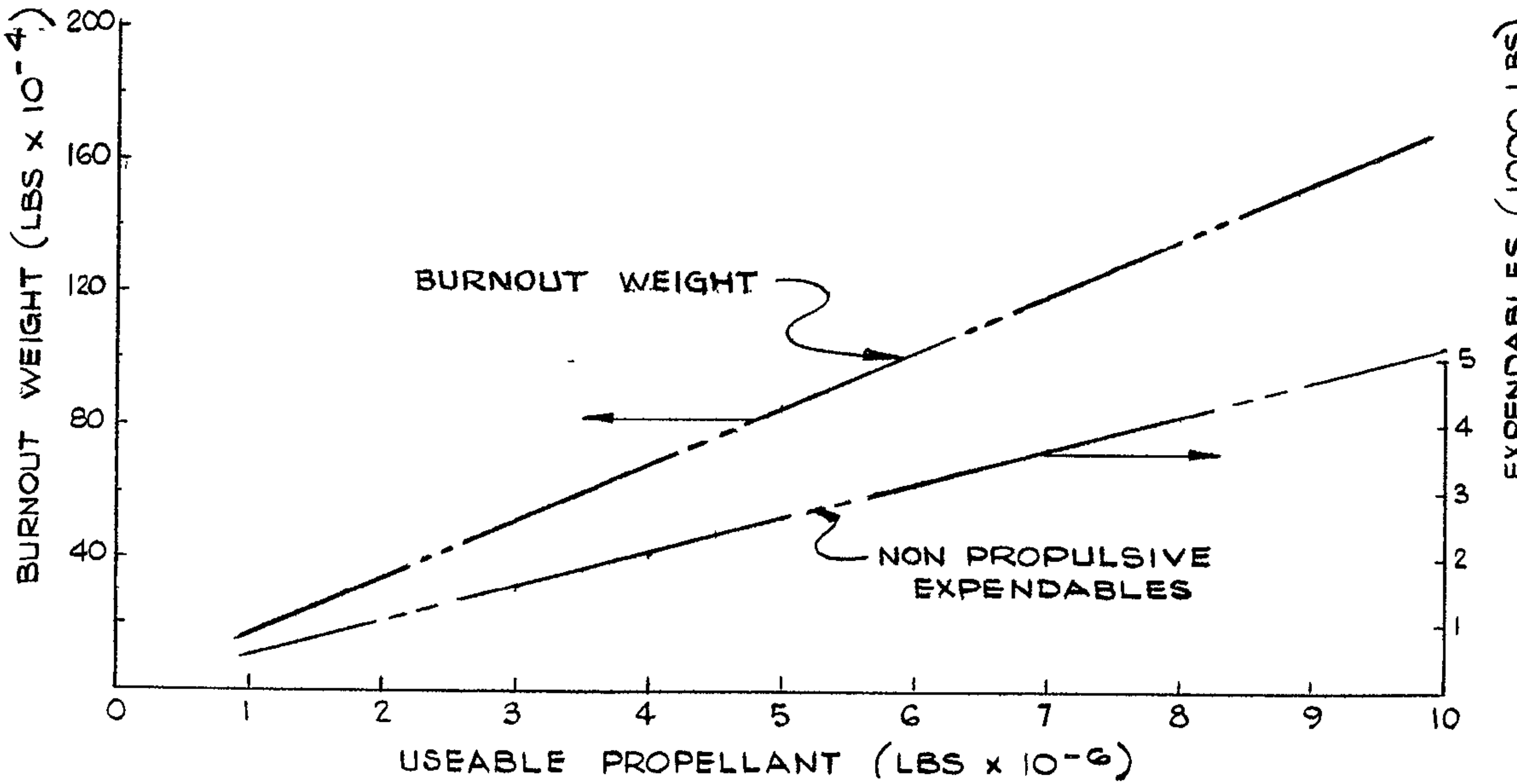


Figure 4-14

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED LOX/RP-1 SECOND STAGE
TWO STAGE CANDIDATE

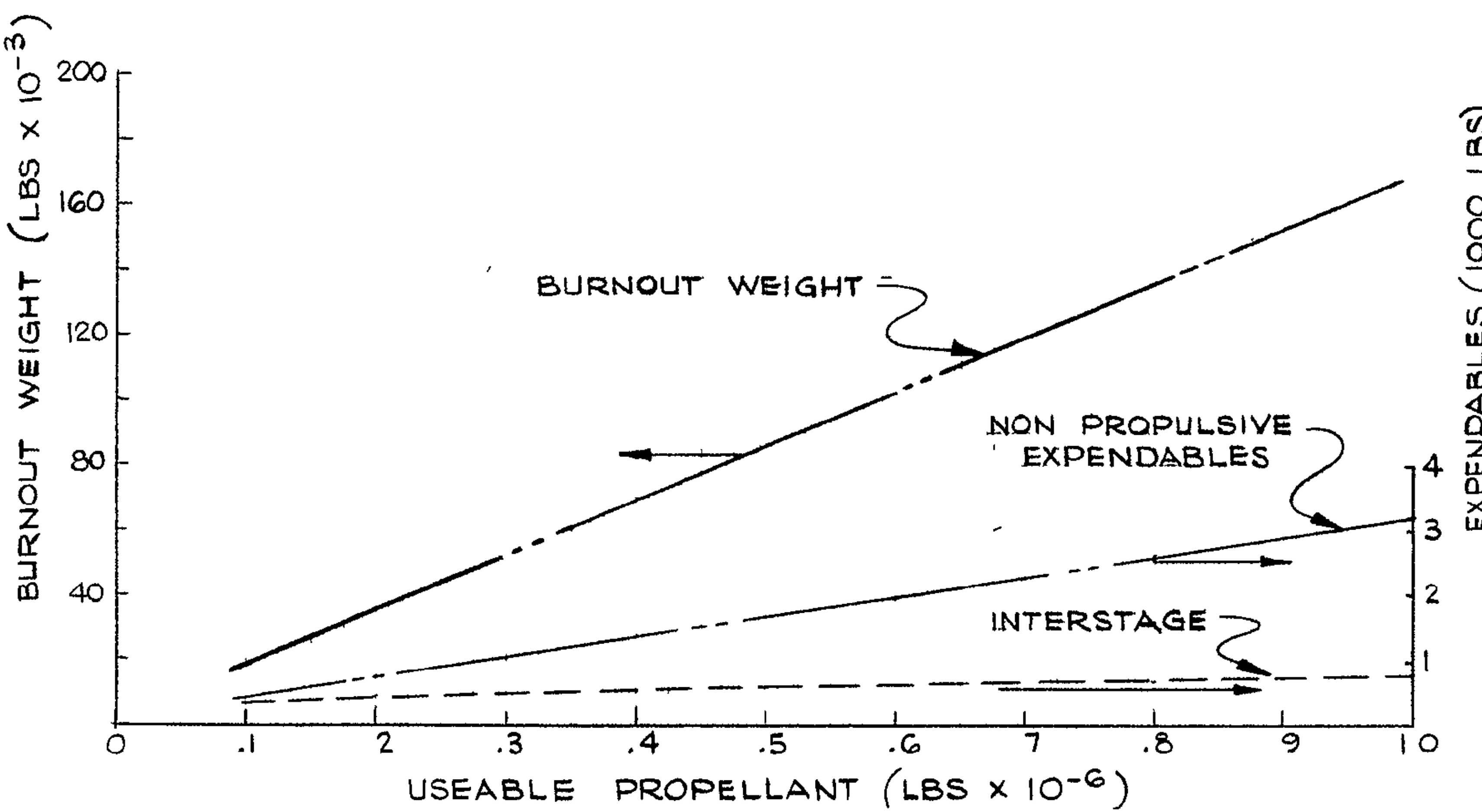


Figure 4-15

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED LOX/RP-1 FIRST STAGE
THREE STAGE CANDIDATE

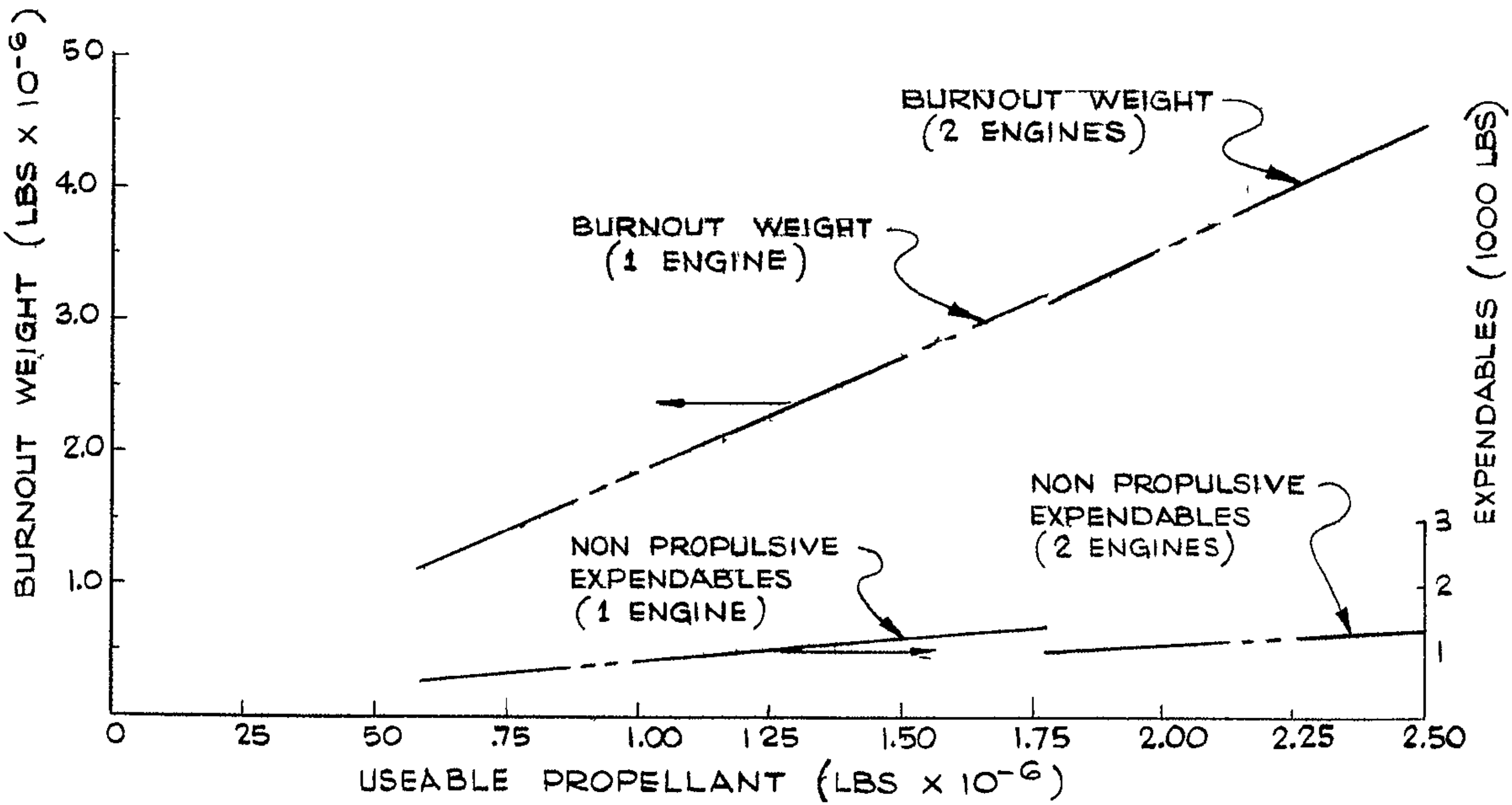


Figure 4-16

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED LOX/RP-1 SECOND STAGE
THREE STAGE CANDIDATE

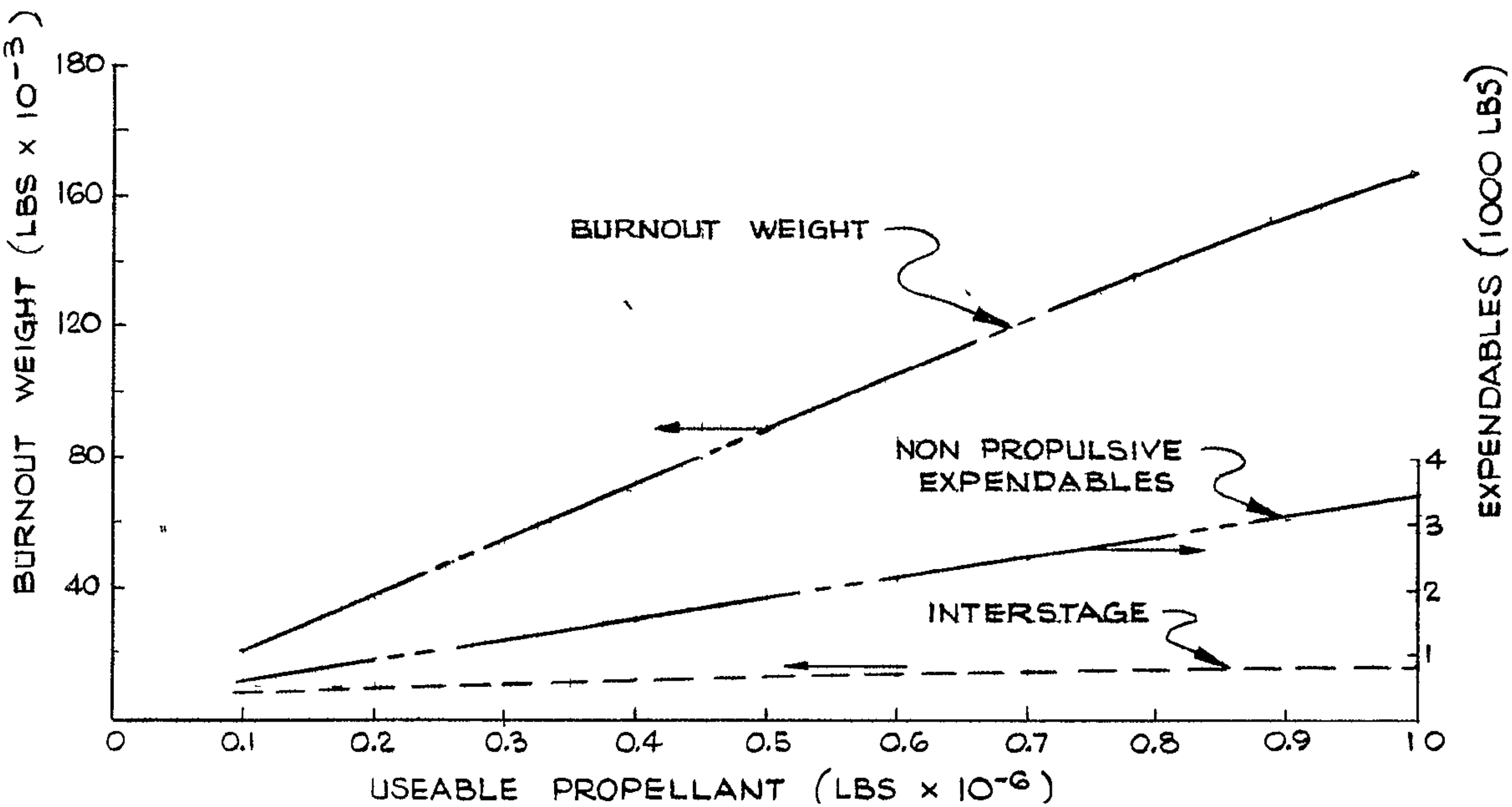


Figure 4-17

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED LOX/RP-1 THIRD STAGE
THREE STAGE CANDIDATE

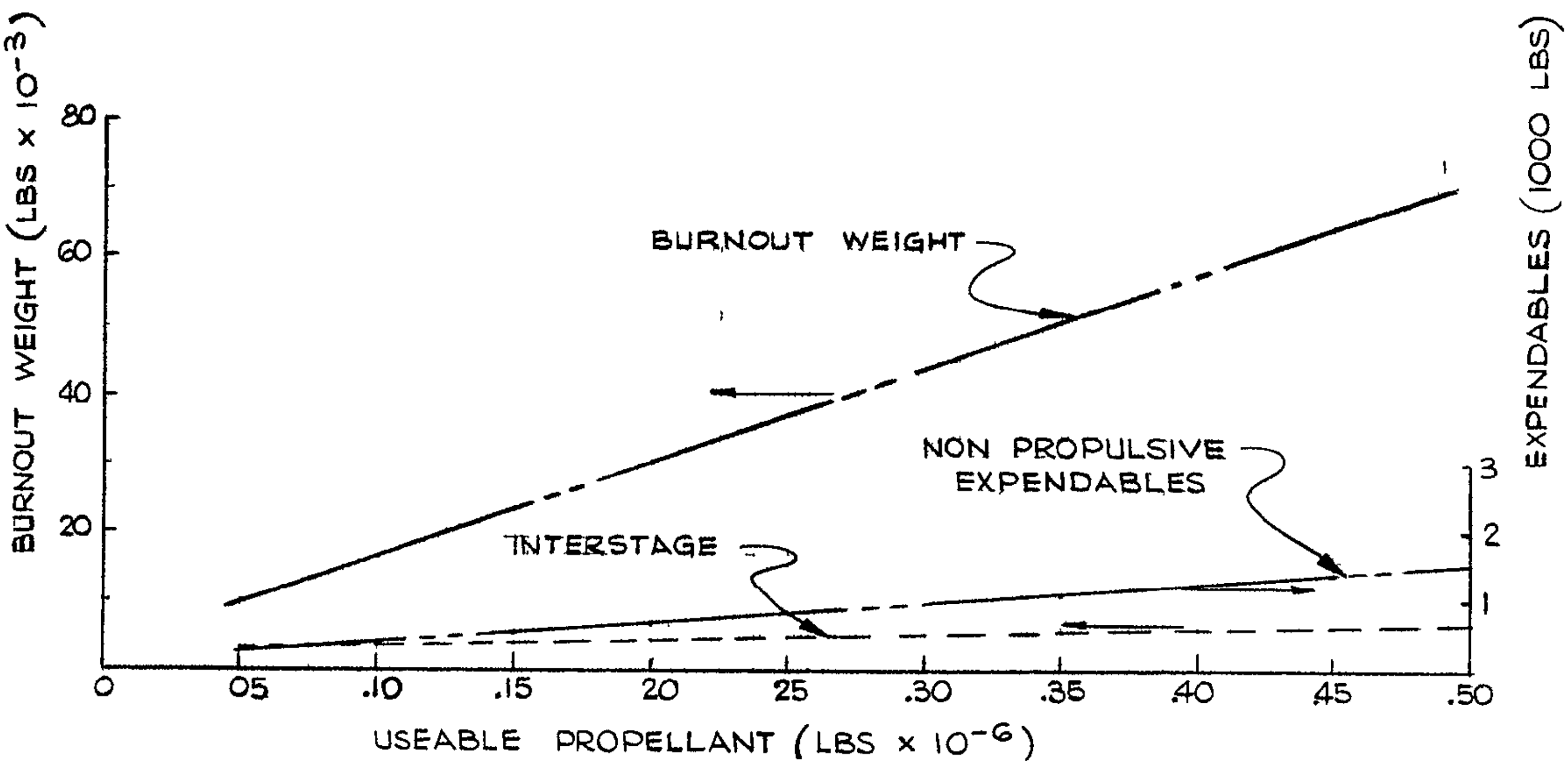


Figure 4-18

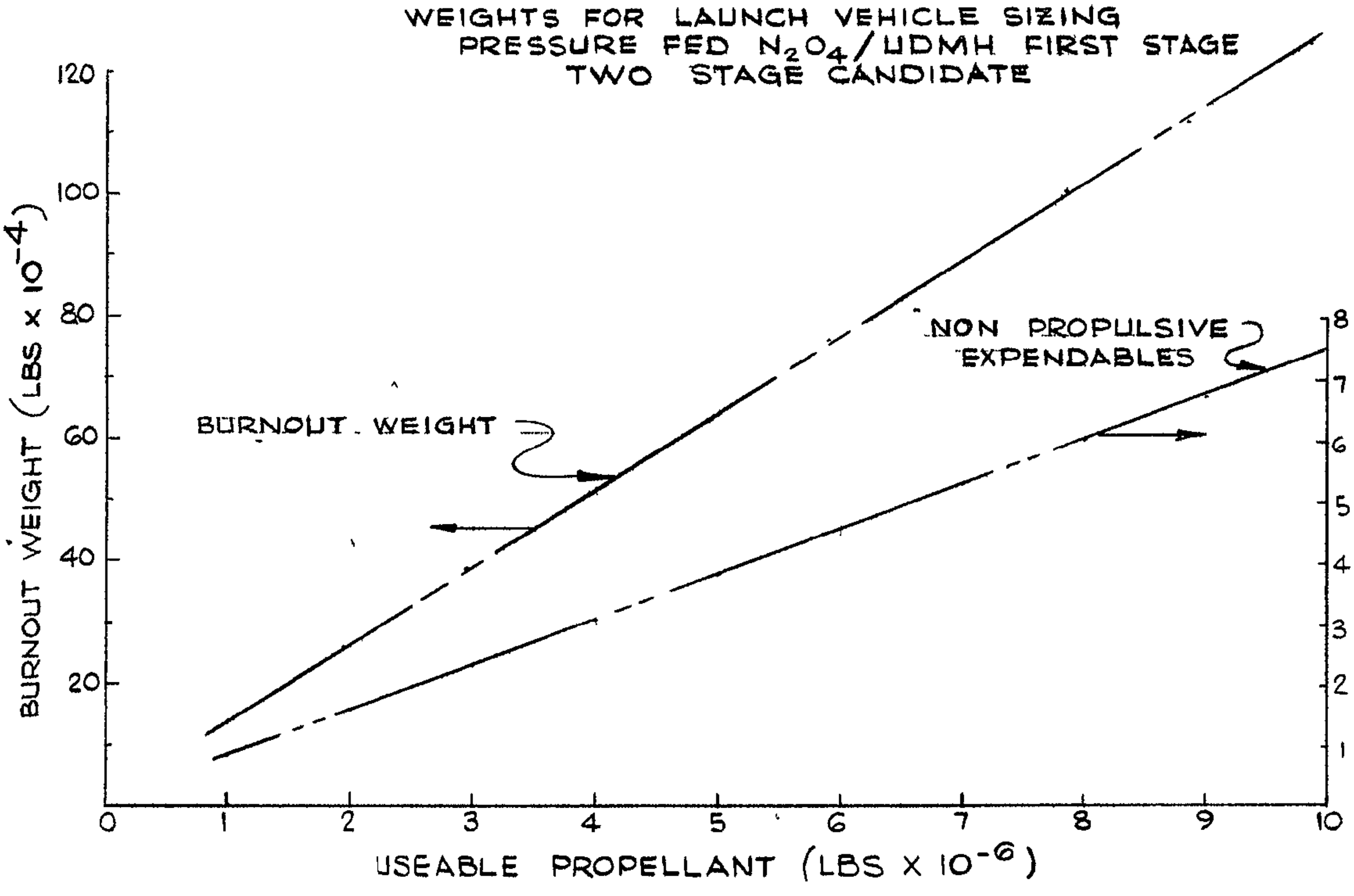

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED N_2O_4/UDMH FIRST STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10^{-4})
EXPENDABLES (1000 LBS)
USEABLE PROPELLANT (LBS x 10^{-6})
BURNOUT WEIGHT
NON PROPULSIVE EXPENDABLES
0
1
2
3
4
5
6
7
8
9
10
20
40
60
80
100
120

Figure 4-19

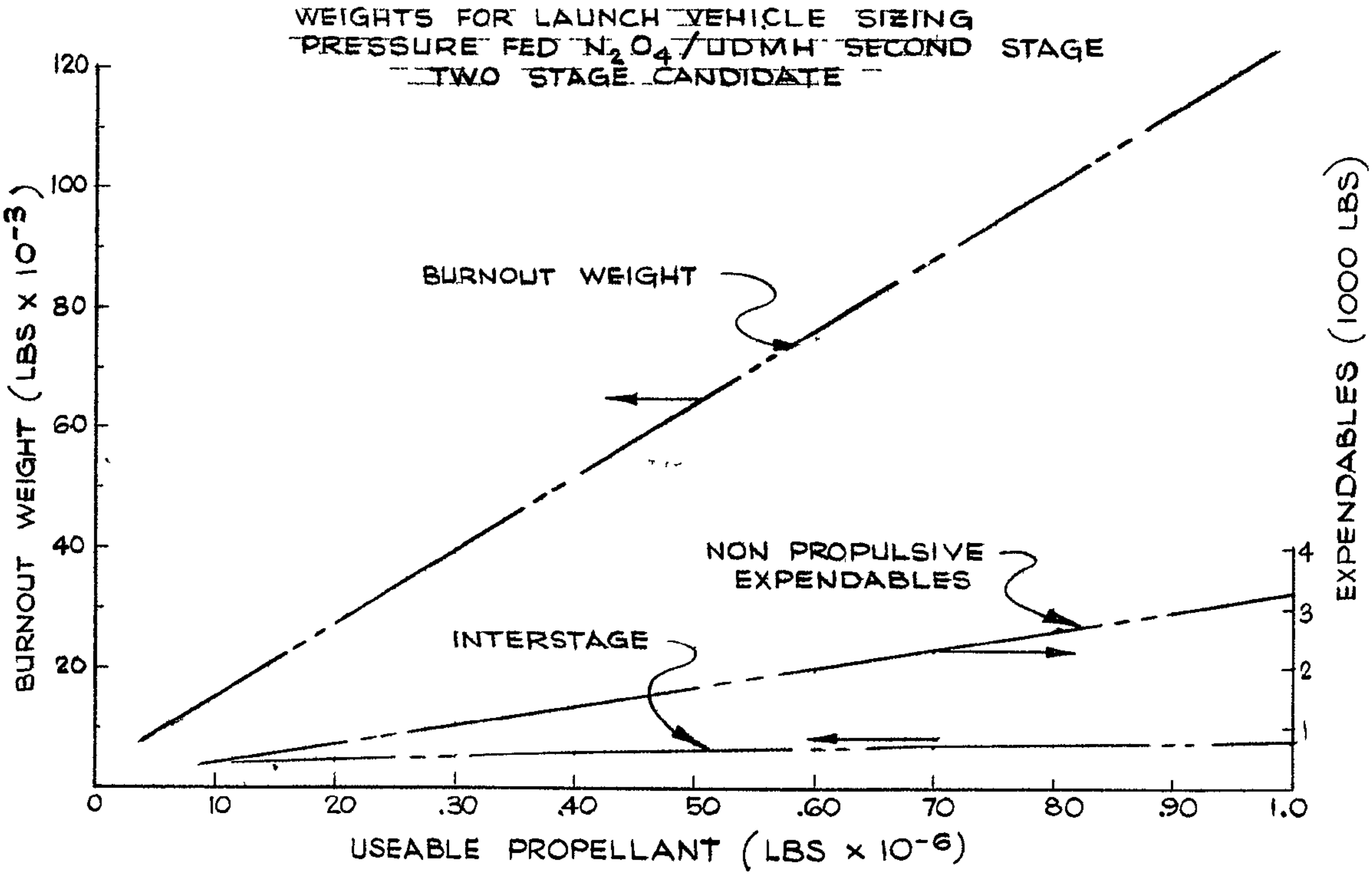

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED N2O4/UDMH SECOND STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT
NON PROPULSIVE
EXPENDABLES
INTERSTAGE
BURNOUT WEIGHT (LBS x 10-3)
EXPENDABLES (1000 LBS)
USEABLE PROPELLANT (LBS x 10-6)
120
100
80
60
40
20
0
10
20
.30
.40
50
.60
70
80
.90
1.0
1
2
3
4

Figure 4-20

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED N_2O_4/UDMH FIRST STAGE
THREE STAGE CANDIDATE

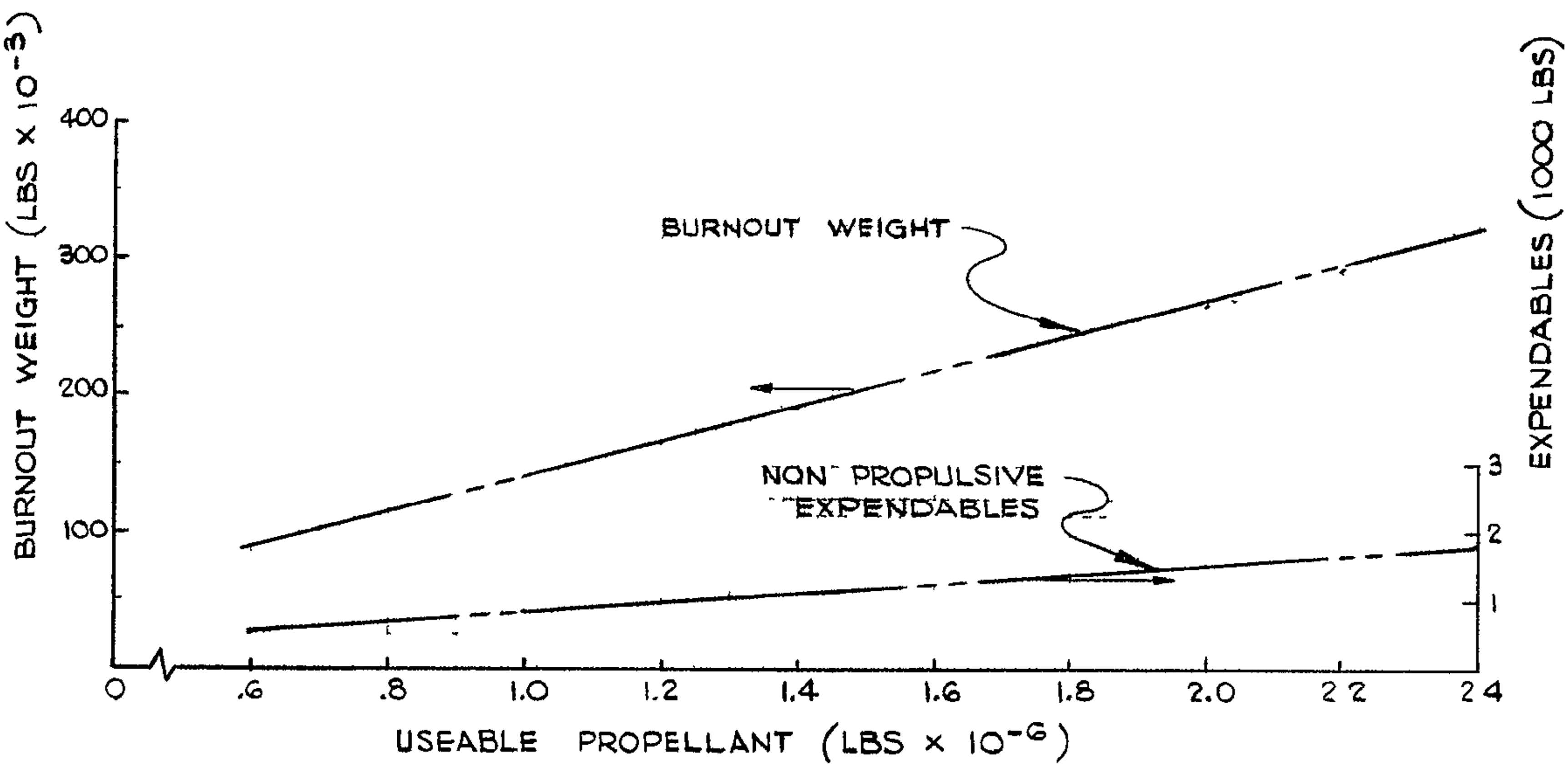


Figure 4-21

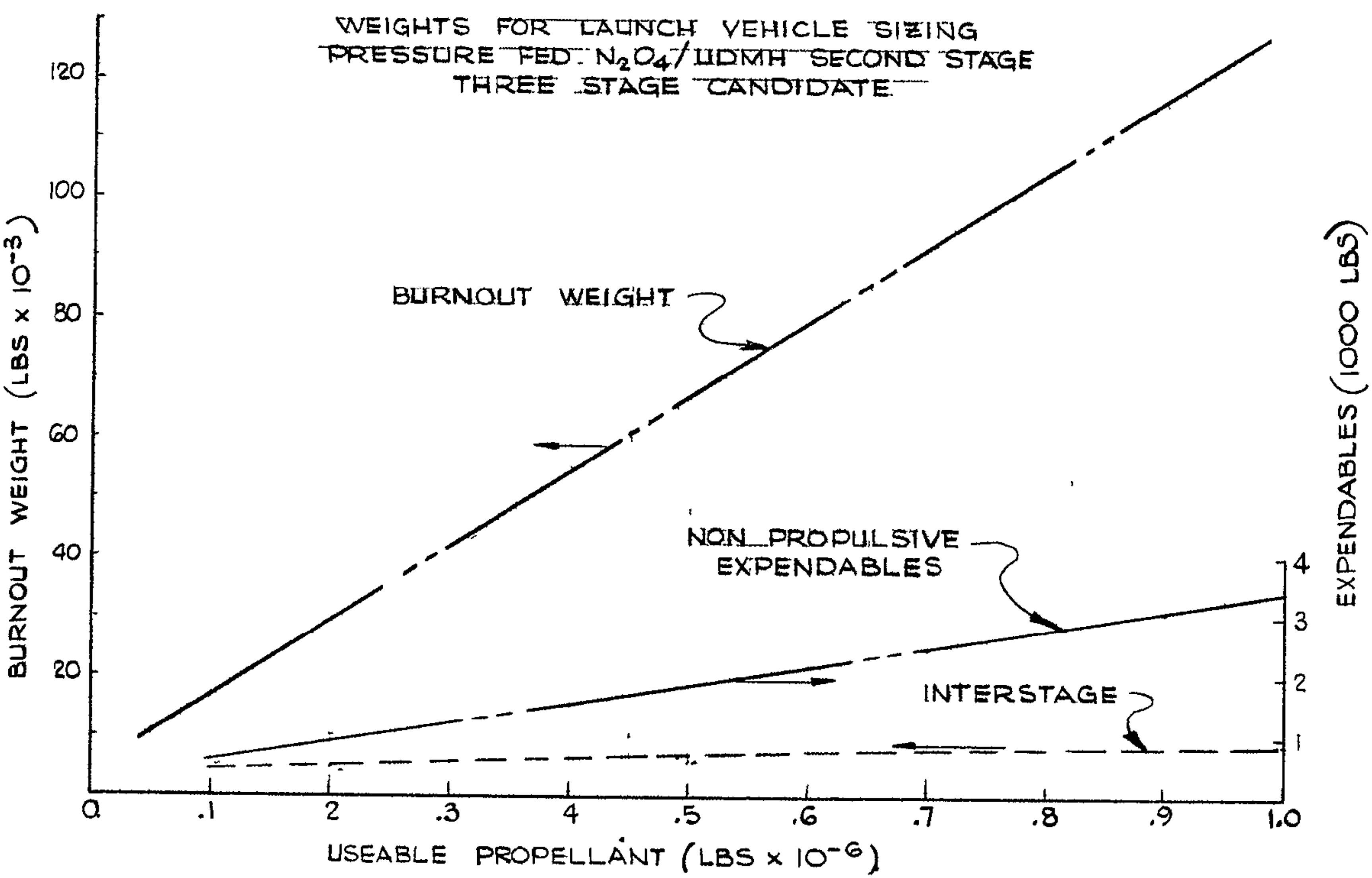

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED N_2O_4/UDMH SECOND STAGE
THREE STAGE CANDIDATE
BURNOUT WEIGHT
NON PROPULSIVE EXPENDABLES
INTERSTAGE
BURNOUT WEIGHT (LBS x 10^{-3})
120
100
80
60
40
20
EXPENDABLES (1000 LBS)
4
3
2
1
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
USEABLE PROPELLANT (LBS x 10^{-6})

Figure 4-22

WEIGHTS FOR LAUNCH VEHICLE SIZING
PRESSURE FED N_2O_4/UDMH THIRD STAGE
THREE STAGE CANDIDATE

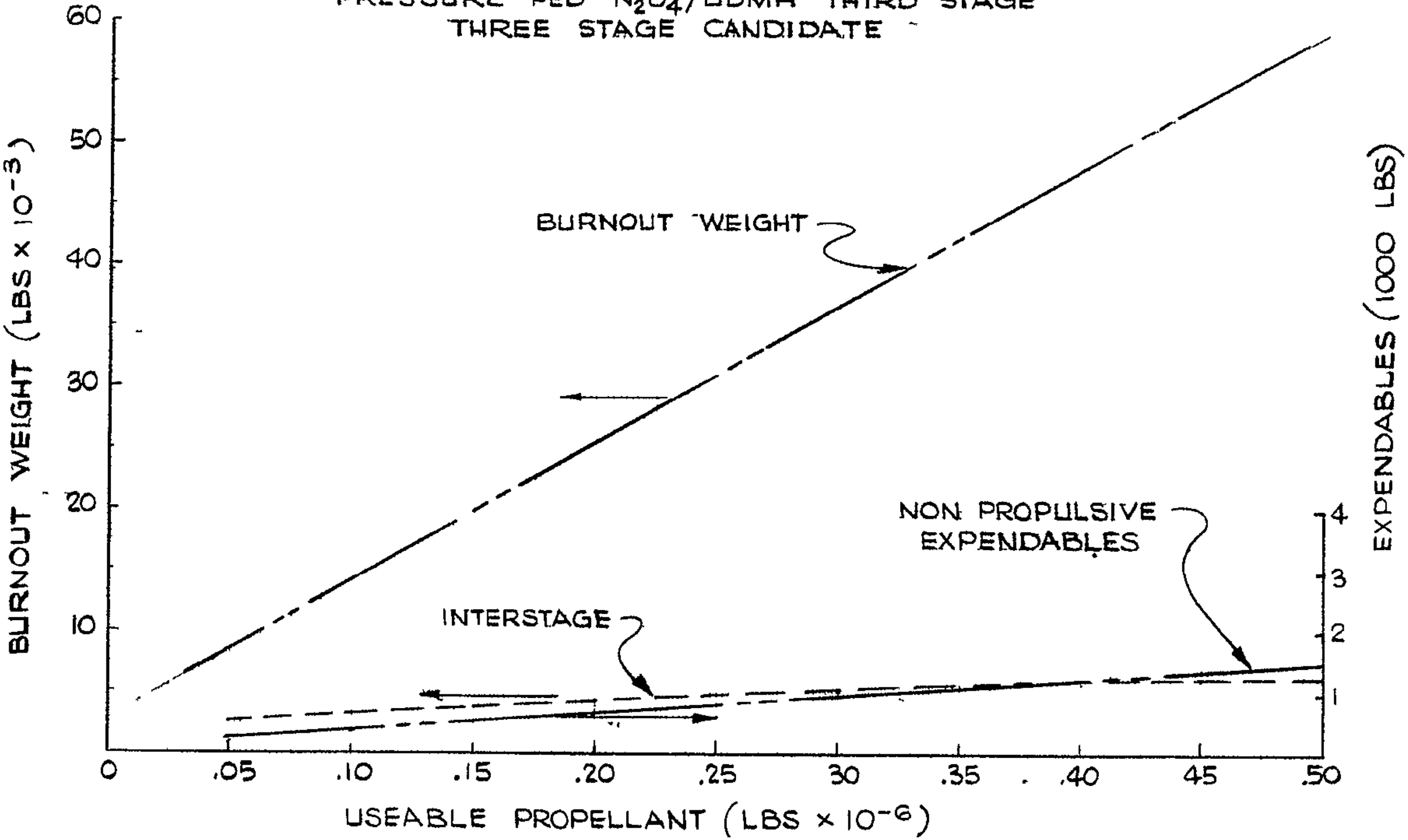


Figure 4-23

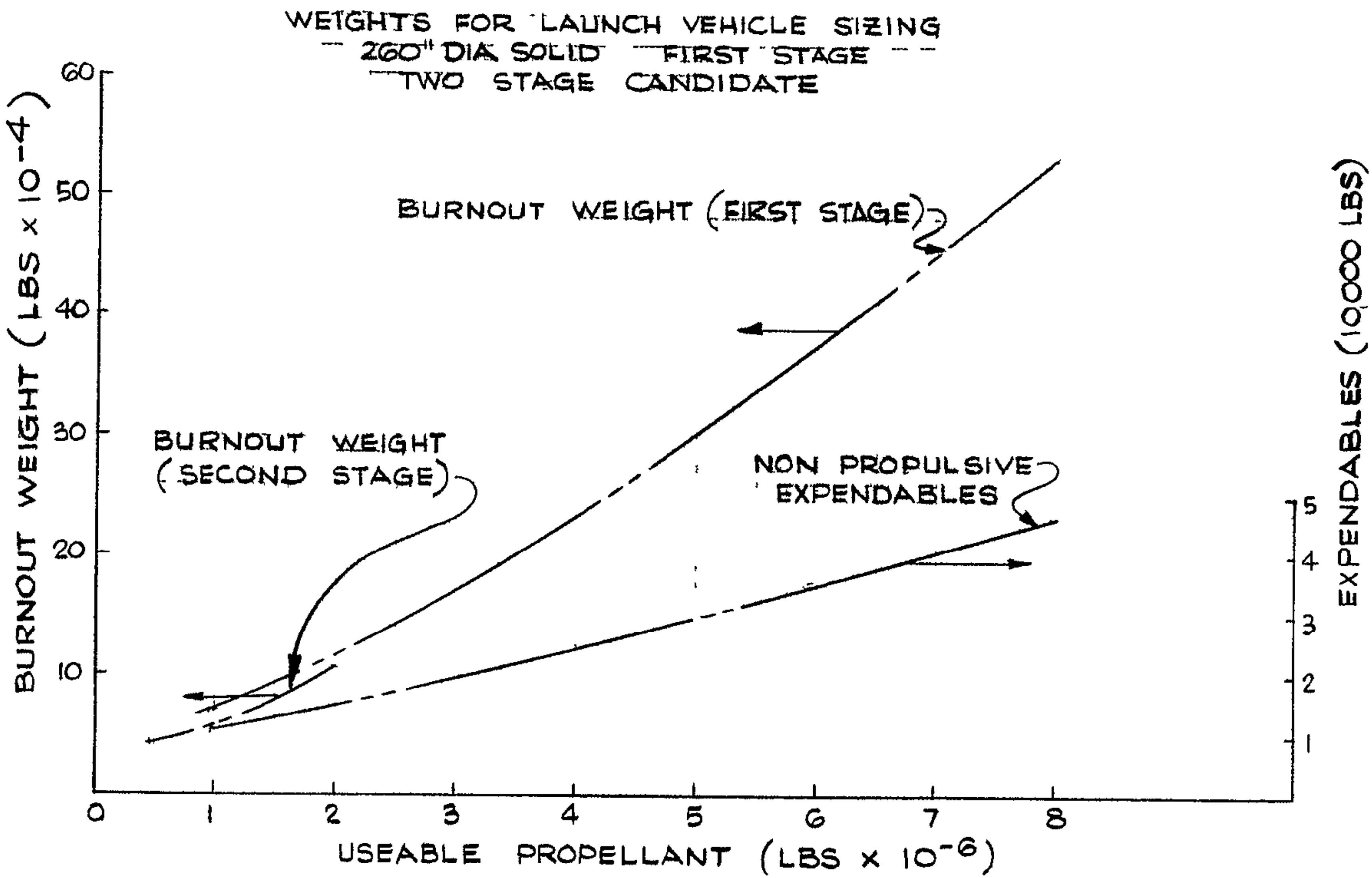

WEIGHTS FOR LAUNCH VEHICLE SIZING
260" DIA SOLID FIRST STAGE
TWO STAGE CANDIDATE
BURNOUT WEIGHT (LBS x 10^-4)
60
50
40
30
20
10
BURNOUT WEIGHT (FIRST STAGE)
BURNOUT WEIGHT (SECOND STAGE)
NON PROPULSIVE EXPENDABLES
EXPENDABLES (10,000 LBS)
5
4
3
2
1
0
1
2
3
4
5
6
7
8
USEABLE PROPELLANT (LBS x 10^-6)

Figure 4-24

4 2 Parametric Cost Analysis - Cost was estimated as a function of size of each of the 32 launch vehicle concepts that were analyzed Several precautions were used to assure the estimates were compatible First, before the cost of any vehicle was calculated, the cost categories and definitions (discussed below) were defined for all vehicles Second, during the estimating procedure, all assumptions and methodologies were continuously examined to assure they were uniformly applied to all vehicles and did not artifically bias the results for or against any vehicle Third, the basic cost data from which the estimates were derived, was analyzed to verify that the design objectives, service requirements and program assumptions for the subject system were philosophically equivalent to the assumptions used for pricing competitive systems The lowest cost available was used only if it was accompanied by sufficiently detailed information to identify exactly what the cost included (See below for further discussions of low cost) Fourth, the factors used for scaling and modifying existing cost data were held uniform from vehicle to vehicle Fifth, emphasis was placed on accurately evaluating the differences between vehicles rather than detail costing of each minor part of each vehicle With these precautions, the resulting vehicle costs were not necessarily the minimum cost and may not always reflect the absolute cost of the system, but their relative values were compatible and a high level of confidence can be placed in the relative costs of the vehicle systems

The parametric costs are estimated to have an overall accuracy of approximately ±25% However the relative accuracy between vehicles is estimated to be ±10% The final phase of the study would have refined both the design and cost of the vehicles to permit accurate choice between those vehicles whose parametric costs differed by less than 10%

The costing itself was done in two phases During the first phase, the recurring costs of the vehicle hardware were determined These costs were expressed parametrically in terms of the dollars per pound of propellant in the stages of each vehicle and were used as inputs to the sizing analysis During the second phase, the launched cost of the specific vehicles selected in the sizing analysis were determined The launched cost was computed by adding the propellant and launch crew cost to the hardware cost The hardware costs were determined by using the parametric costs that were developed in the first phase

The propellant costs were based on the delivered cost of the propellants and the size of the vehicle tanks The launch crew sizes were determined by a comparison of vehicle operational complexities and differences Austere program philosophy, a frozen design, no catchup work at launch, and four launches per year were assumed

4 2 1 Low Cost Philosophy - The cost estimates reflect a low cost approach This includes (1) simplifying design to permit fewer fabrication and inspection operations, (2) eliminating R&D type instrumentation, (3) minimizing documentation requirements by such techniques as reporting variations only rather than repeating all data on each vehicle, (4) reducing all types of support costs and (5) assuming the vehicle design is frozen and that the vehicle is delivered to the launch site without any open inspection items This low cost approach was incorporated into the study results by deviating from the Saturn R&D type program costs and experience in two significant ways First, the design of the vehicles emphasized lowering of cost and simplifying fabrication rather than optimizing performance Sophisticated systems, such as the present propellant utilization system and the propellant pressurization systems, were replaced by simpler less expensive systems even though the simpler systems slightly degraded vehicle performance. Similarly less efficient engines were assumed because they are easier to fabricate and permit less exacting test procedures Further design economies were provided by using integrally machined skirts This eliminated many pieces and many operations required by the skin and stringer structures that it replaced. Other simplifications and changes are discussed under the section "System Definition "

The second departure from the Saturn program was accomplished by lowering the program support costs and overhead factors This change was incorporated by reducing the fabrication support areas such as sustaining engineering, planning, quality control and logistics Rather than arbitrarily reducing the amount of these support areas, the techniques and cost experience achieved in the Thor-Delta program were substituted into the Saturn based cost estimating procedure Since these support areas are manned at a lower level in the Thor-Delta operational program than in the present Saturn R&D program, this change provided some cost reduction A true minimum cost program might permit a further reduction in these support and documentation requirements However, since no data is available to establish the ultimate minimum support costs, the Thor/Delta experience was used

4 2 2 First Unit Costs - The basic cost estimates were made for the first production unit of each subsystem Determining the cost of the first production unit eliminated the variations that arise when different learning curves are used A further simplifying assumption was to ignore any test units that might be built as part of the RDT&E phase before the first production unit These assumptions eliminated some detail refinements without degrading the compatibil·ty of the resulting estimates

4 2 3 Learning Curves - After the hardware cost estimates were completed for all vehicles, a learning curve was applied in two ways In the first approach, an approximate average cost for the first 40 production vehicles was determined by multiplying the first unit cost by a factor of 0 57 This factor is the average unit cost of the first forty units on a 90% Wright learning curve The cost of the launch operations and the propellant in the loaded vehicle was added to this hardware cost to derive the total launched cost of the vehicles This first method ignored the learning curve was the same vehicles This introduces a slight error Therefore, a second, more accurate learning curve adjustment was applied to the five top launch vehicle candidates selected during the sizing analysis portion of the study A composite learning curve was calculated for each stage of the vehicle These costs were used to compute the vehicle quantity cost curve shown in Figure 4-25

4 2.4 Cost Methodology and Category Definitions - The cost of each vehicle was computed for the structure, engine, astrionics, and attitude control systems Tables 4-2, 4-3 and 4-4 show typical results for three vehicles A fifth subsystem, pressurization system was added for some vehicles Whenever a vehicle included solid-rocket motors, the cost of the motor case, nozzle, and propellant was included in lieu of the engine cost The cost of each subsystem was calculated for different propellant weights These subsystem costs were then added to give stage hardware costs at different propellant weights

The structural cost includes the cost of the propellant tanks (except solid-rocket motor cases), skirts, interstage, and engine thrust structure The cost of astronics support structure was included in the astronic equipment

The structural portion of the cost estimate for each stage was derived by calculating the costs for the skirts, tanks, thrust structure, and interstage as defined for the peg-point design of that stage The costs for the skirts, tanks, and thrust structure were added to obtain the stage structural cost These were converted to dollars per pound of structure, and the costs for similar stages of different weights were computed by using scaling factors

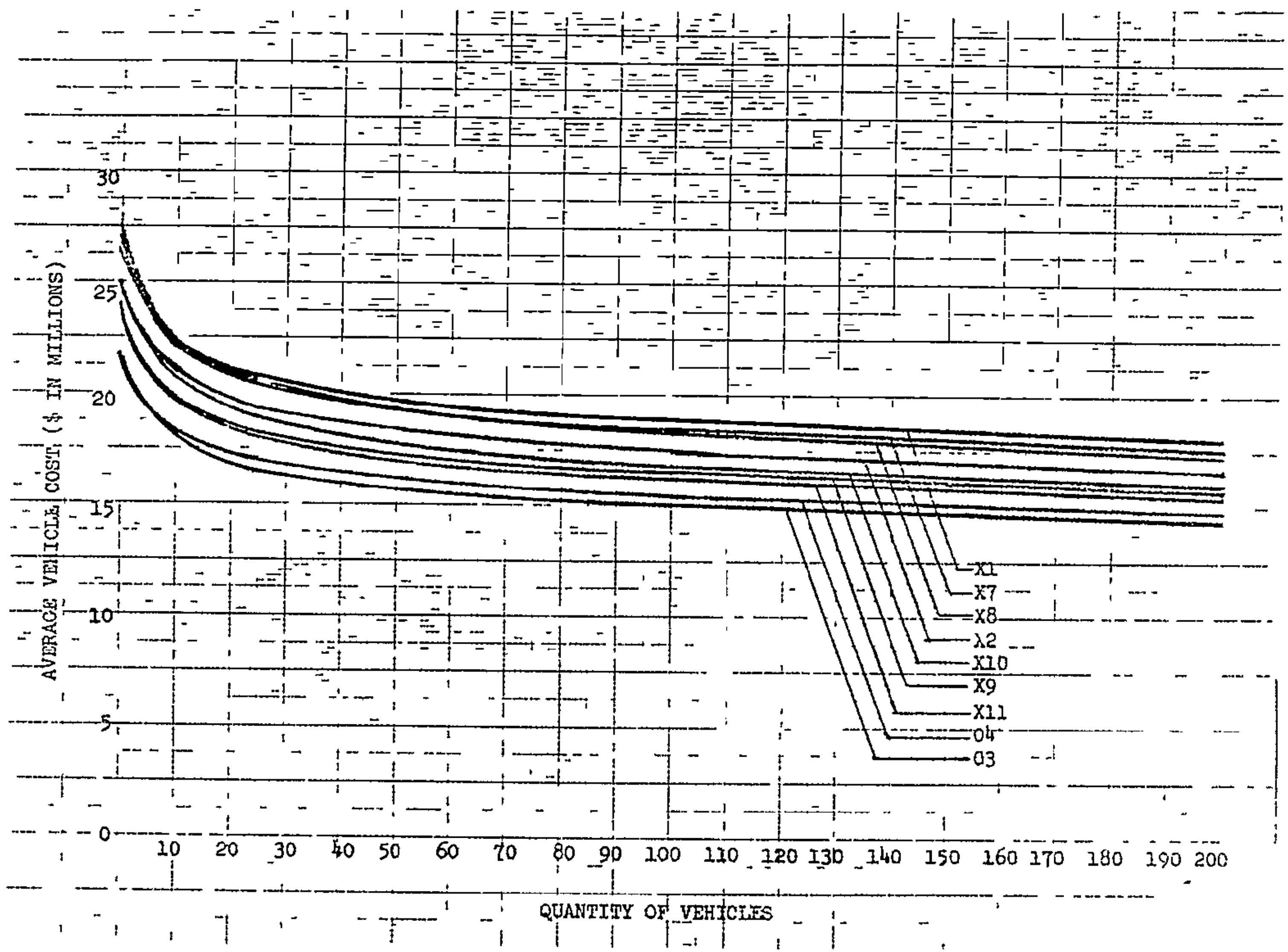


Figure 4-25 Launched Cost vs Quantity of Vehicles

Table 4-2

COST BREAKDOWN FOR SELECTED VEHICLE THREE-STAGE SOLID, SOLID, LO_2/RP-1 FIRST UNIT COST ($ Millions)

	Stage I	Stage II	Stage III	Total
Type	Solid	Solid	LO_2/RP-1	
W_p	1,991,000	901,000	437,000	
Structure	$ 0 6	$ 1 8	$ 3 0	$ 5 4
SRM	$ 4 3	$ 2 0	---	$ 6 3
Engine	---	---	$ 0 7	$ 0 7
TVC	$ 0 5	$ 0 3	$ 0 2	$ 1 0
Avionics	$ 1 2	$ 1 2	$ 1 8	$ 4 2
	$ 6 6	$ 5 3	$ 5 7	$ 17 6

Table 4-3

COST BREAKDOWN FOR SELECTED VEHICLE THREE-STAGE SOLID, SOLID, LO_2/LH_2 FIRST UNIT COST ($ Millions)

	Stage I	Stage II	Stage III	Total
Type	Solid	Solid	LOX/LH_2	---
W_p	2,859,000	535,000	87,000	---
Structures	$ 0 7	$ 1 1	$ 1 8	$ 3 6
SRM	$ 5 9	$ 1 3	---	$ 7 2
Engine	---	---	$ 1 4	$ 1 4
TVC	$ 0 7	$ 0 2	$ 0 1	$ 1 0
Avionics	$ 1 2	$ 1 2	$ 2 2	$ 4 6
Total	$ 8 5	$ 3 8	$ 5 5	$ 17 8

Table 4-4

COST BREAKDOWN FOR SELECTED VEHICLE TWO STAGE. SOLID, LO_2/LH_2 FIRST UNIT COST ($ Millions)

	Stage I	Stage II	Total
Type	Solid	LOX/LH_2	---
W_p	2,400,000	408,000	---
Structures	0.7	5.4	6 1
SRM	5.1	---	5 1
Engine	---	4 2	4.2
TVC	0 7	0.3	1 0
Avionics	1 2	2.6	3.8
	7 7	12 5	20 2

The use of a composite structural cost as described above was required because a structural weight breakdown was not available for any stage size except the peg-point To use this composite structural figure it was necessary to assume that the weight relationship between the tanks, skirts, and engine thrust structure of any given type of stage did not change significantly as the stage varied in size This assumption introduces a negligible error whenever the final selected stage size is close to the peg-point size (Note One of the tasks in the phases of the study that was cancelled would have included refining the cost of the selected vehicle to remove any distortion introduced by this assumption and verify that it did not affect the validity of the final selection)

The estimates for the tank structures were based on S-IVB experience For sump-fed stages, which used a waffle type tank wall, the tank costs were derived from weight scaling the S-IVB costs. For pressure-fed stages, which use a monocoque tank wall, the tank costs were determined by scaling the S-IVB costs with tank pressure as well as weight The cost of conical interstages and engine thrust structure were determined by weight scaling from S-IVB values The cost of cylindrical skirts were determined by weight scaling from S-IVB values The cost of cylindrical skirts were determined by weight scaling from the costs of integrally machined skirts This cost was determined by a detailed Company study where each operation required for this simplified skirt was compared with the time standards developed for the present skirt fabrication operations

The Astrionics subsystem costs includes the electrical power system (batteries, regulators, interconnecting cable), telemetry system (transducers, signal conditioners, multiplexers), control electronics (computer, logic, sequencers, gyros, etc to extent needed), and communications It also includes the electronic logic and sensors that send the signals to such mechanical systems as nozzle actuators, hydraulic systems and control valves It does not include the mechanical systems themselves.

The engine cost estimates were determined separately for each type of engine used by the various vehicles All engine costs were estimated assuming a lower cost engine design with lower performance The cost of the pump-fed LO_2/LH_2 engines were based on the J-2 engine, but the actual cost of the present engine was reduced, based on the figures given in the National Space Booster Study The cost for the pump-fed LO_2/RP-1 engines were based on the F-1 and to a lesser extent the H-1 engine. Based on consultations and data received from engine manufacturers, the engine costs were modified to represent low-cost design Data received from engine manufacturers and comparison with existing engines indicated the difference between pump fed LO_2/RP-1 and pump-fed storable engines were insignificant at the magnitude used in these parametric costs

The pressure-fed LO_2/RP-1 and storable propellant engine costs were derived from combining the nozzle costs for the solid-rocket motors and the chamber-injector costs for the pump-fed liquid engines Had the pressure-fed vehicles ranked as contenders for the optimum vehicles, further analysis of these engine costs would have been desirable. However, the pressure-fed vehicles were sufficiently far from being contenders for minimum cost that a refinement of their engine costs was not necessary

The cost estimates for the solid motors were based on UTC costs for the 120-in. motor, and Aerojet costs for the 260-in motor and Lockheed costs for the 156-in motor Since Lockheed's quotes were lower than the other two, a compromise cost was used This adjusted cost reflected a slightly smaller decrease (approximately 15%) to the Aerojet and UTC cost data than the decrease (approximately 20%) made to the standard engine costs quoted by the liquid-engine companies

The cost estimates for the TVC subsystem was based on the costs determined in a NASA, funded study performed by McDonnell Douglas During this study, various TVC systems were evaluated, normalized, and compared The data for the movable nozzle and liquid injection systems were readily adaptable to the systems specified in this study

Cost estimates for the pressurization system were made only if a major subsystem were required The pressurization system for most of the vehicles consisted of a gas bleed from the engines The cost of the lines, regulators, and valves required for this system was too small to influence the choice between vehicles On the other hand, the LO_2/RP-1 pressure-fed vehicles required a solid-propellant gas generator pressurization system which was separately costed The cost of this system was based on the cost of small rocket motors Similarly, the cost of the bipropellant liquid gas generator used for the storable pressure-fed vehicles was calculated It was based on the cost determined for the bipropellant roll control system used for the solid boosted S-IVB A third pressurization system was specified for the LO_2/RP-1 pump-fed stages The cost for this blowdown system was calculated by computing the cost of the pressure tanks required Since the cost of the lines, valves, etc were not specifically included in the cost of most of the stages, no cost for them was included when the pressurization system was itemized for those stages requiring a major subsystem

Using the category definition and cost methodology described in the preceding paragraphs, Figure 4-26 through 4-47 show the specific costs of various vehicle stages plotted against propellant load

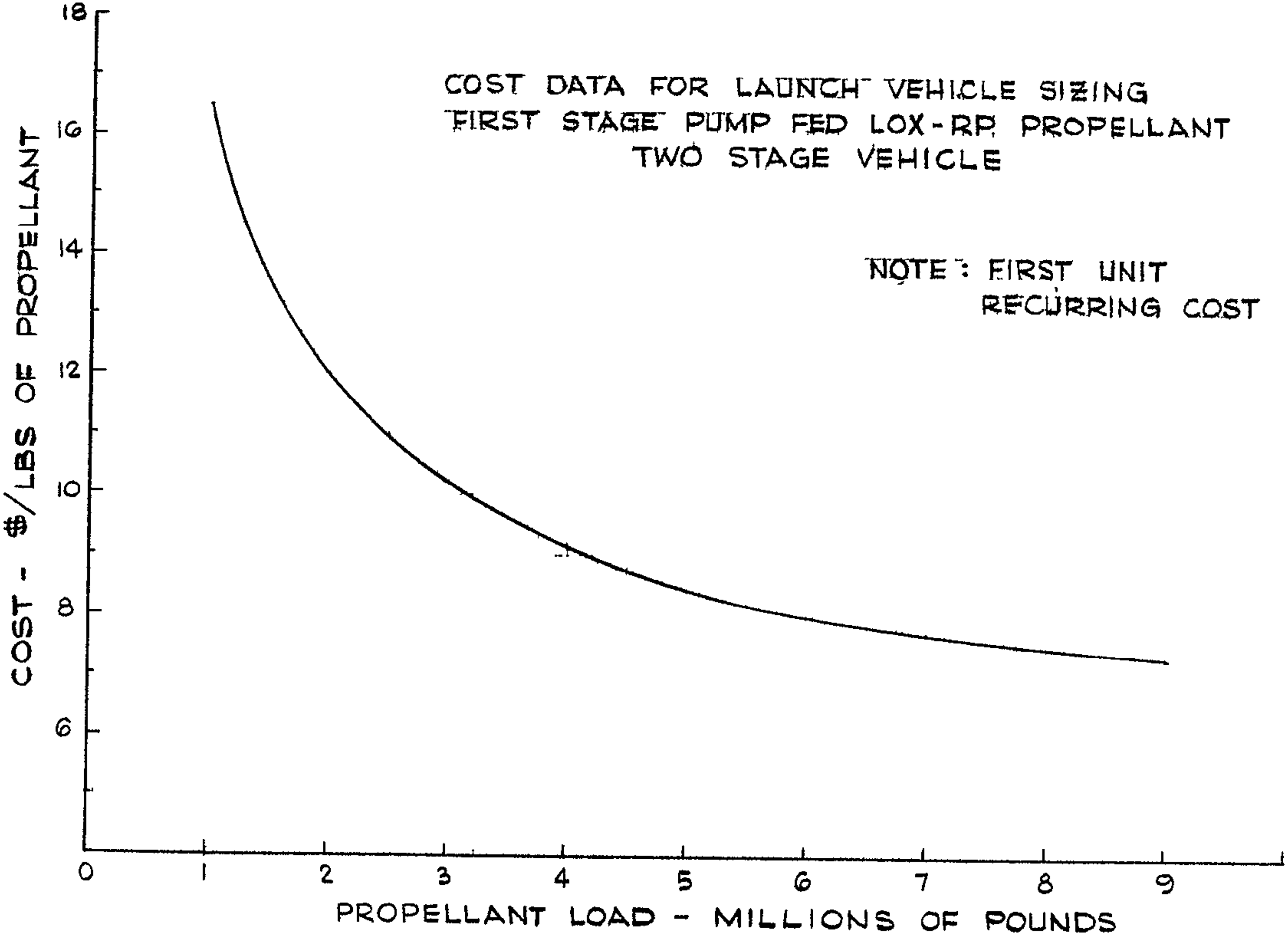

COST DATA FOR LAUNCH VEHICLE SIZING
FIRST STAGE PUMP FED LOX-RP PROPELLANT
TWO STAGE VEHICLE
NOTE: FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
18
16
14
12
10
8
6
0
1
2
3
4
5
6
7
8
9
PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-26

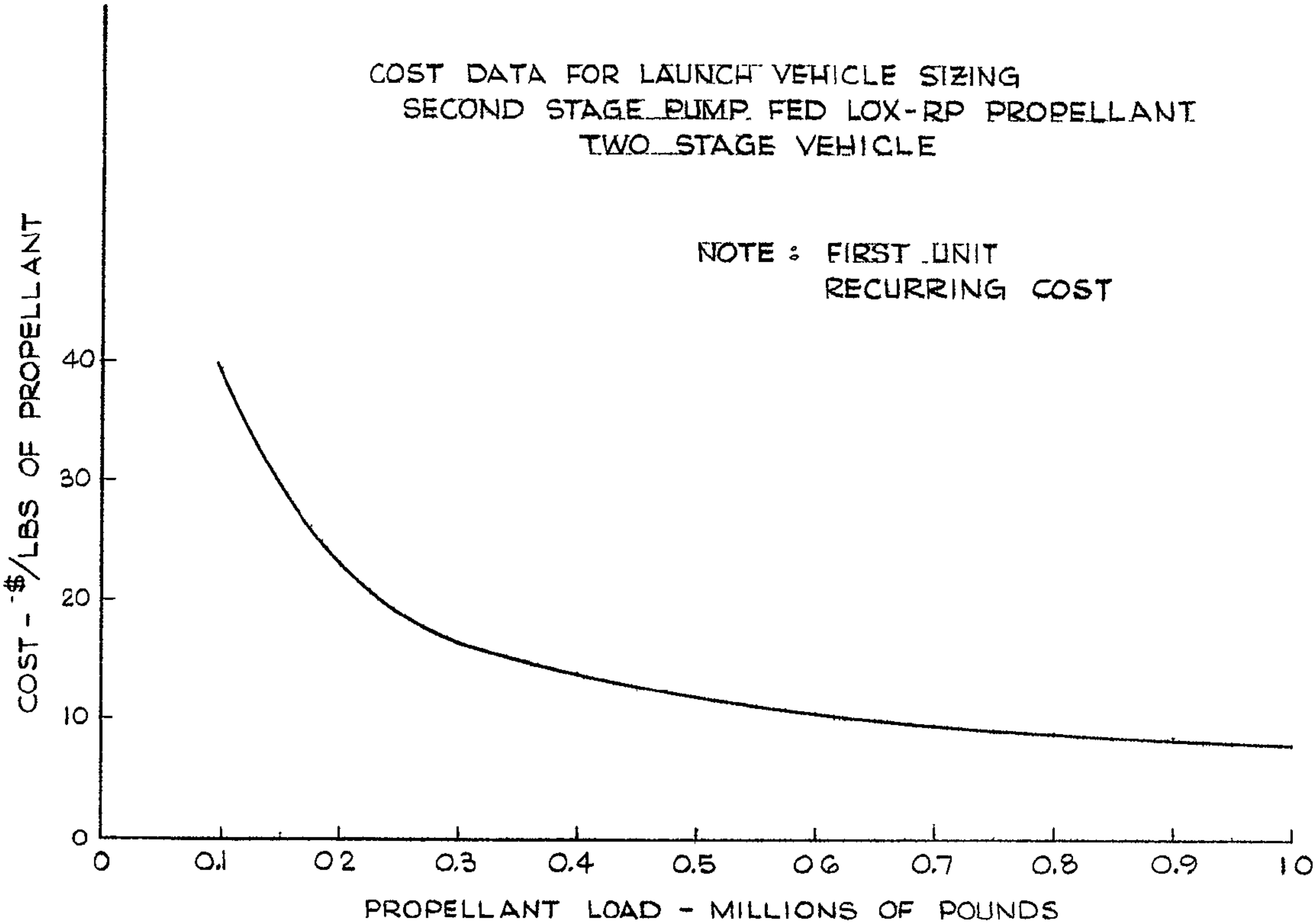

COST DATA FOR LAUNCH VEHICLE SIZING
SECOND STAGE PUMP FED LOX-RP PROPELLANT
TWO STAGE VEHICLE
NOTE : FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
0
10
20
30
40
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-27

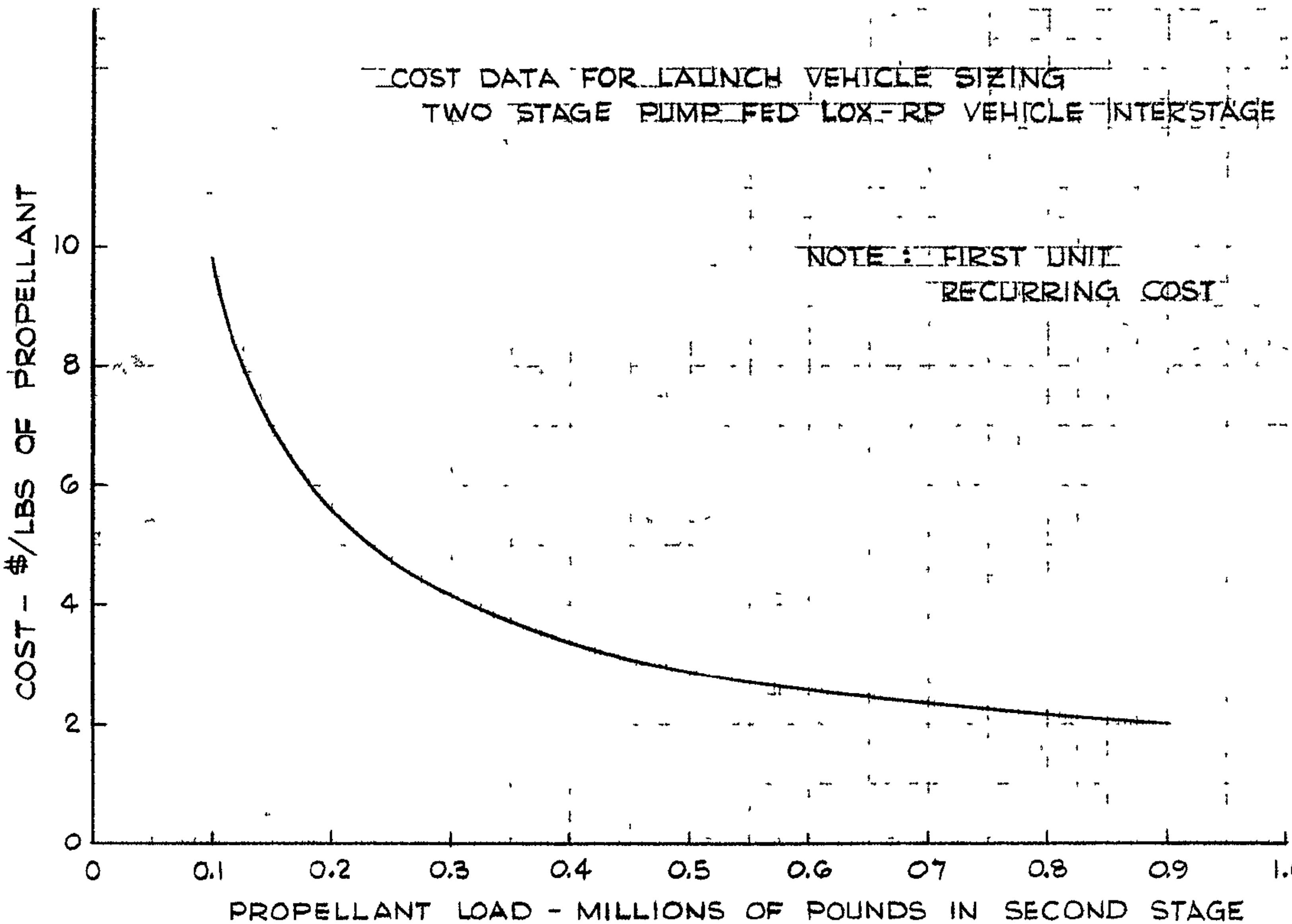

COST DATA FOR LAUNCH VEHICLE SIZING
TWO STAGE PUMP FED LOX-RP VEHICLE INTERSTAGE
NOTE: FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
0
2
4
6
8
10
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
PROPELLANT LOAD - MILLIONS OF POUNDS IN SECOND STAGE

Figure 4-28

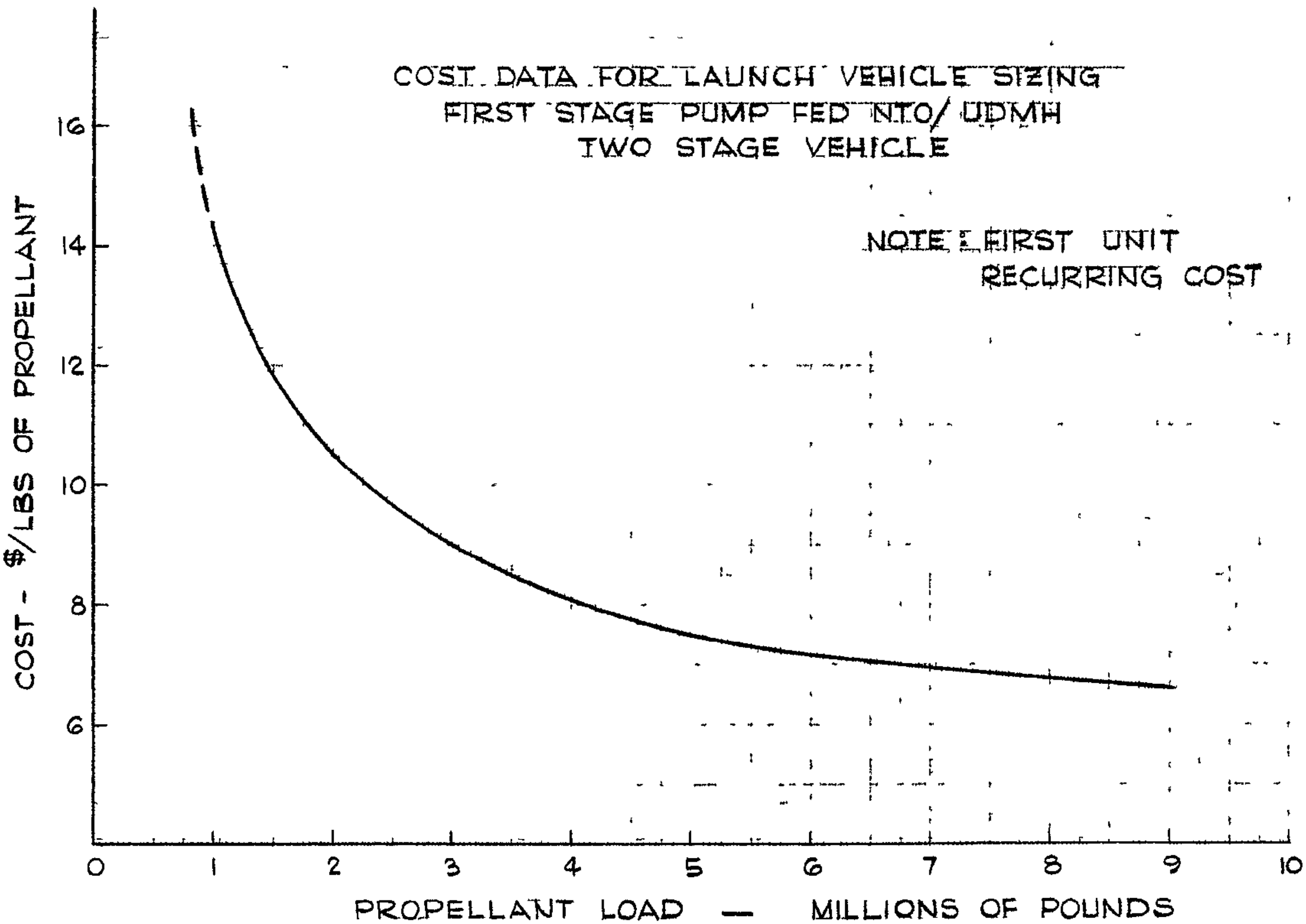

COST DATA FOR LAUNCH VEHICLE SIZING
FIRST STAGE PUMP FED NTO/UDMH
TWO STAGE VEHICLE
NOTE: FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
16
14
12
10
8
6
0
1
2
3
4
5
6
7
8
9
10
PROPELLANT LOAD — MILLIONS OF POUNDS

Figure 4-29

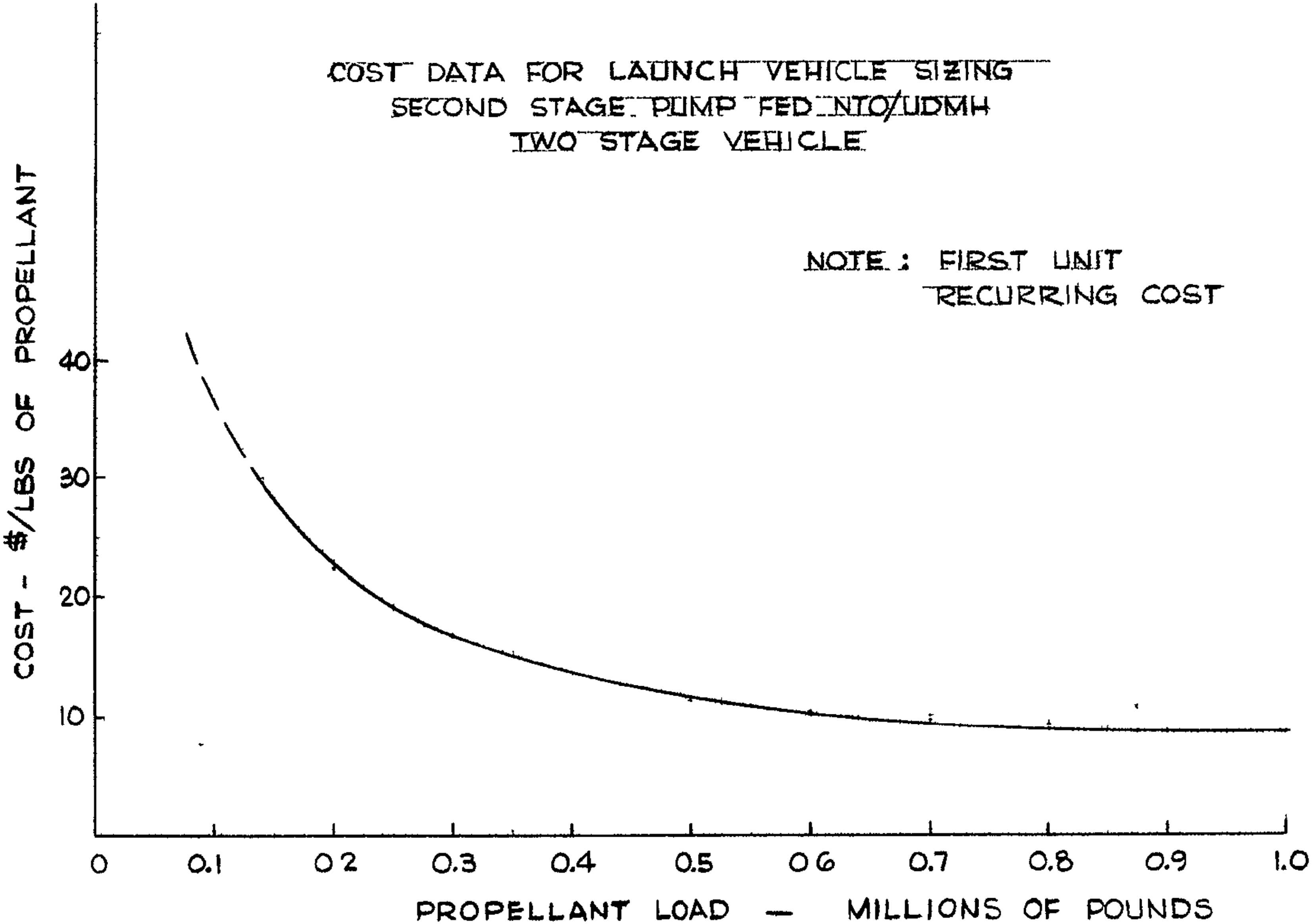

COST DATA FOR LAUNCH VEHICLE SIZING
SECOND STAGE PUMP FED NTO/UDMH
TWO STAGE VEHICLE
NOTE: FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
40
30
20
10
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
PROPELLANT LOAD — MILLIONS OF POUNDS

Figure 4-30

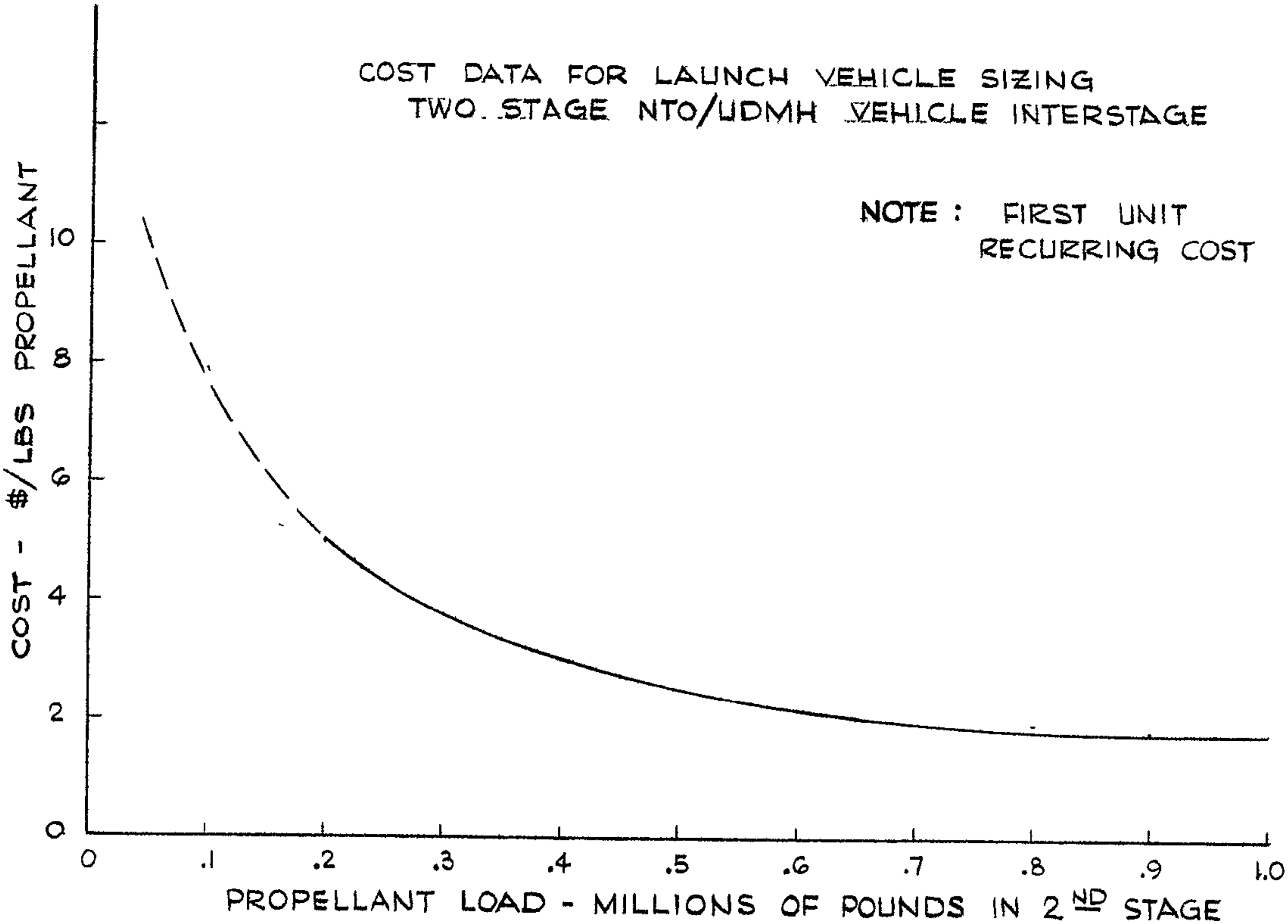

COST DATA FOR LAUNCH VEHICLE SIZING
TWO STAGE NTO/UDMH VEHICLE INTERSTAGE
NOTE : FIRST UNIT
RECURRING COST
COST - $/LBS PROPELLANT
0
2
4
6
8
10
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
PROPELLANT LOAD - MILLIONS OF POUNDS IN 2ND STAGE

Figure 4-31

COST DATA FOR LAUNCH VEHICLE SIZING
FIRST STAGE PUMP FED LOX-RP
THREE STAGE VEHICLE

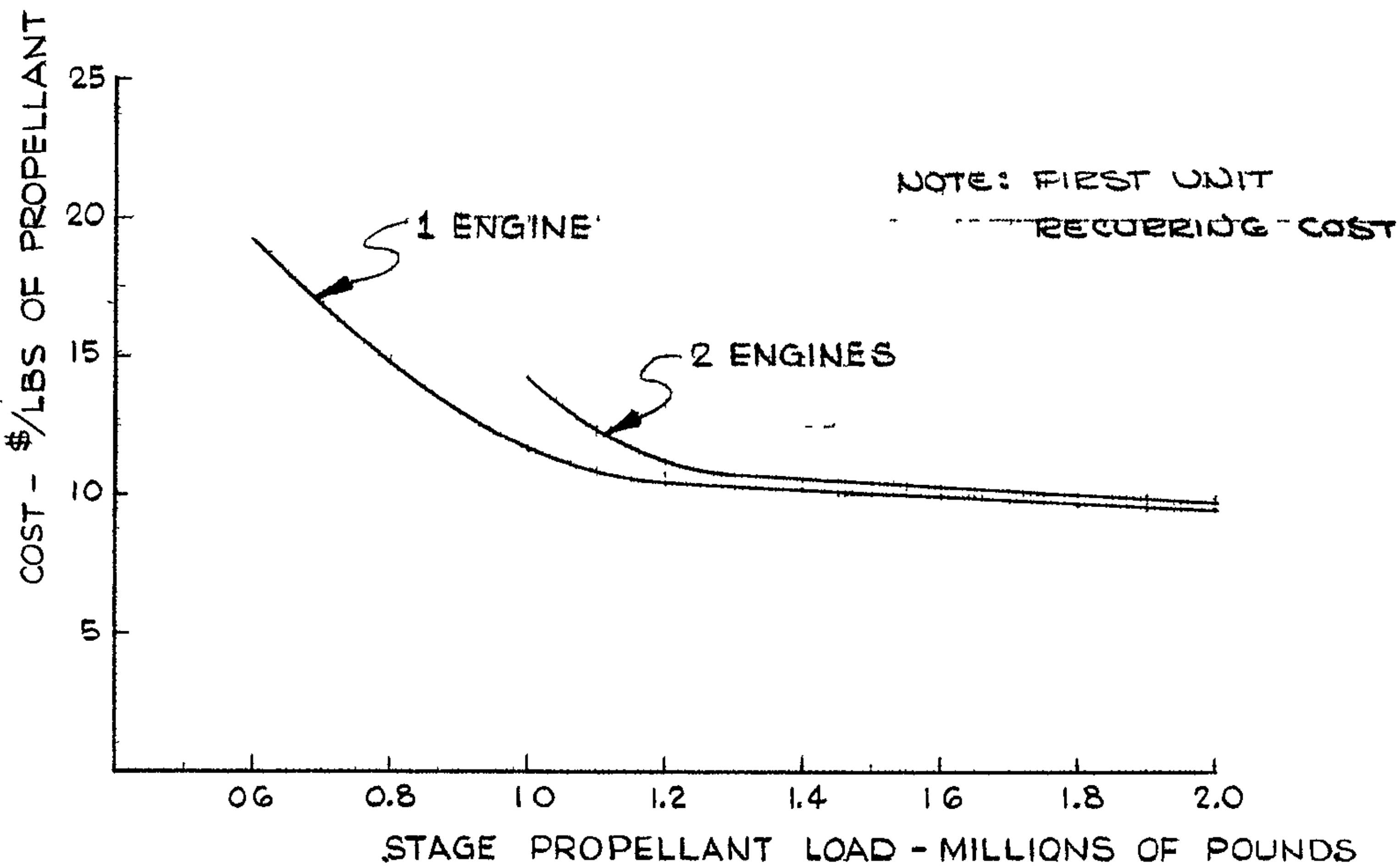


Figure 4-32

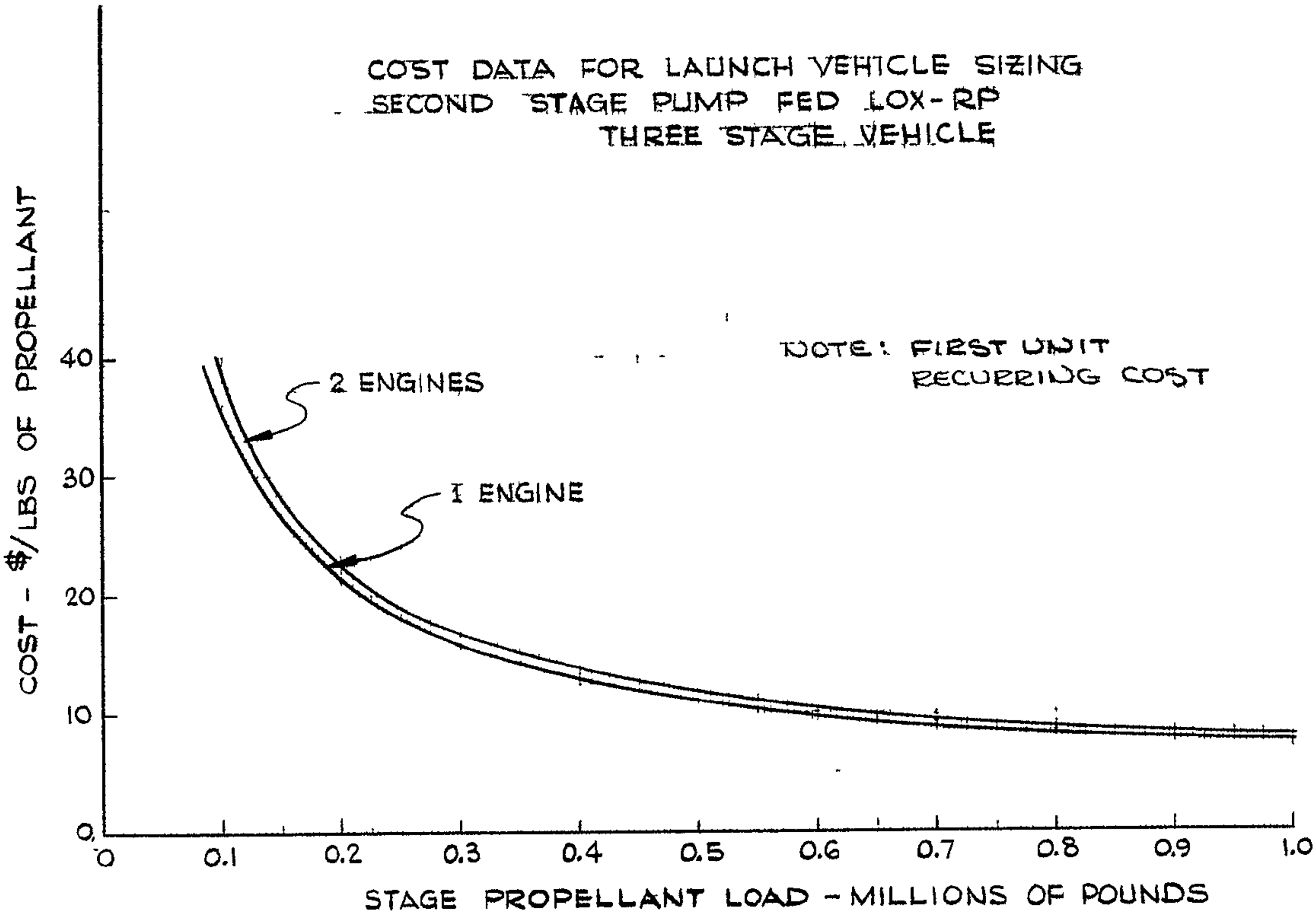

COST DATA FOR LAUNCH VEHICLE SIZING
SECOND STAGE PUMP FED LOX-RP
THREE STAGE VEHICLE
NOTE: FIRST UNIT
RECURRING COST
2 ENGINES
1 ENGINE
COST - $/LBS OF PROPELLANT
0
10
20
30
40
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-33

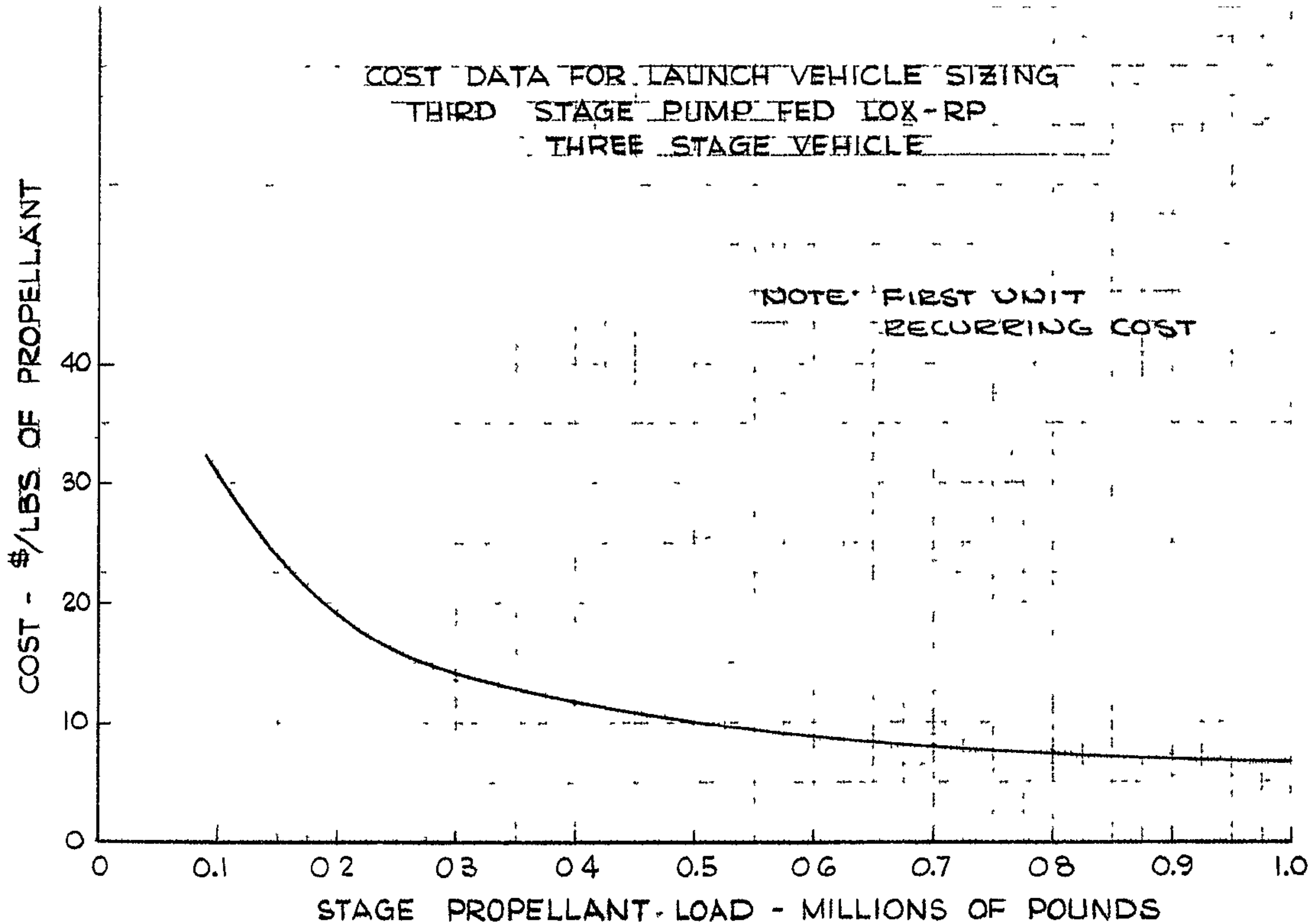

COST DATA FOR LAUNCH VEHICLE SIZING
THIRD STAGE PUMP FED LOX-RP
THREE STAGE VEHICLE
NOTE: FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
0
10
20
30
40
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-34

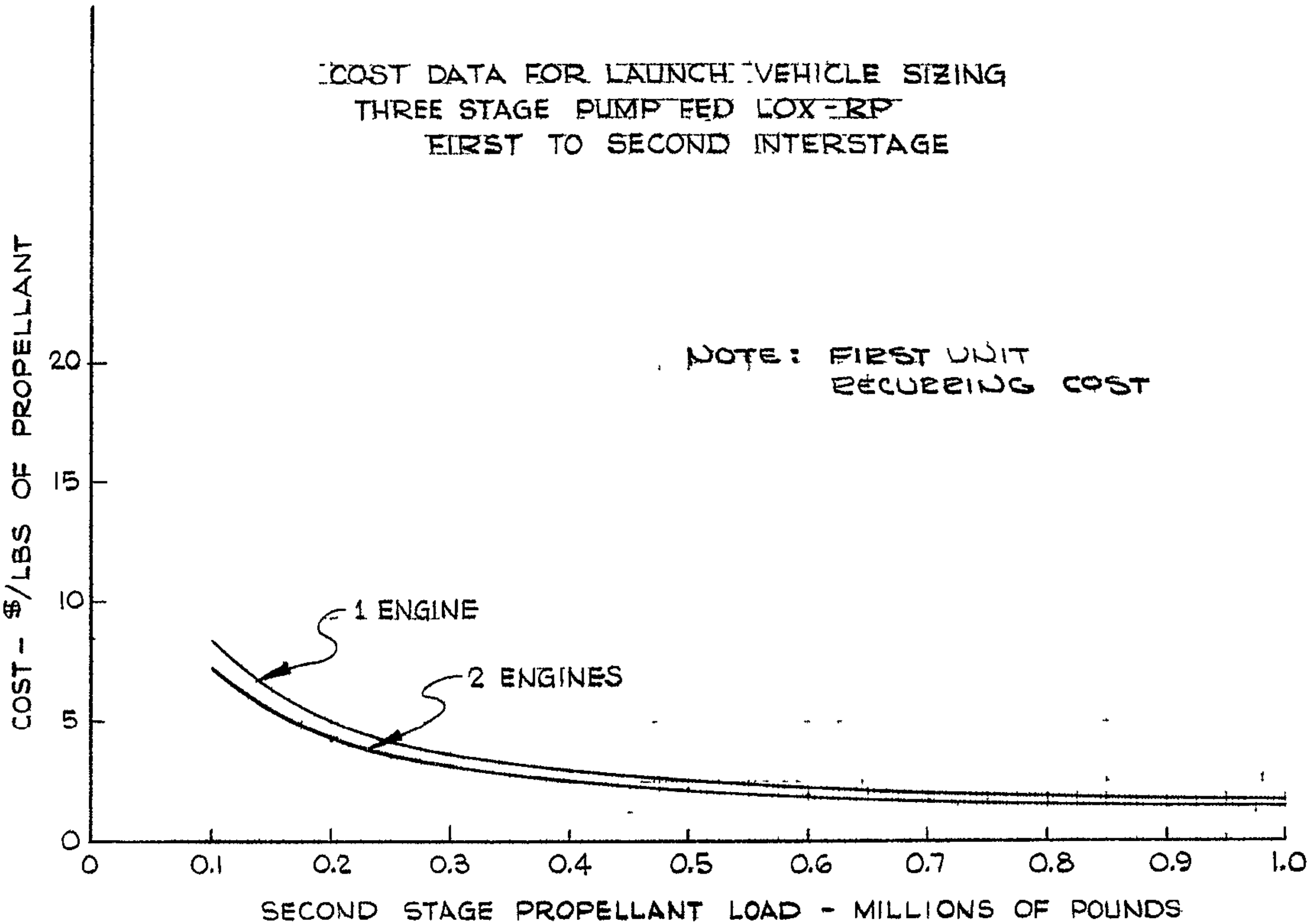

COST DATA FOR LAUNCH VEHICLE SIZING
THREE STAGE PUMP FED LOX-RP
FIRST TO SECOND INTERSTAGE
NOTE: FIRST UNIT
RECURRING COST
1 ENGINE
2 ENGINES
COST - $/LBS OF PROPELLANT
0
5
10
15
20
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
SECOND STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-35

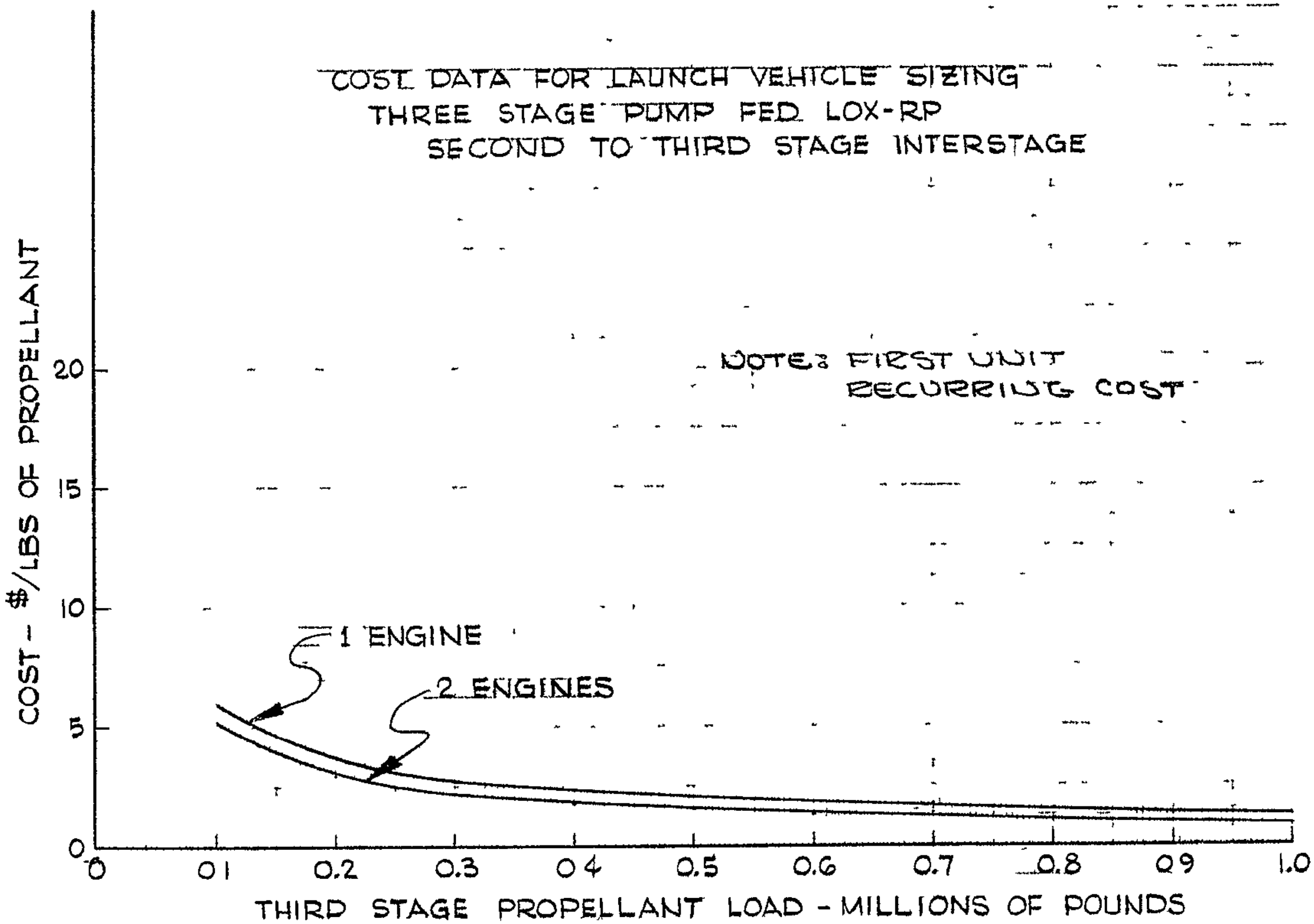

COST DATA FOR LAUNCH VEHICLE SIZING
THREE STAGE PUMP FED LOX-RP
SECOND TO THIRD STAGE INTERSTAGE
NOTE: FIRST UNIT
RECURRING COST
COST - $/LBS OF PROPELLANT
20
15
10
5
0
1 ENGINE
2 ENGINES
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
THIRD STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-36

COST DATA FOR LAUNCH VEHICLE SIZING
FIRST STAGE STORABLE PUMP FED (X 32)
THREE STAGE CANDIDATE

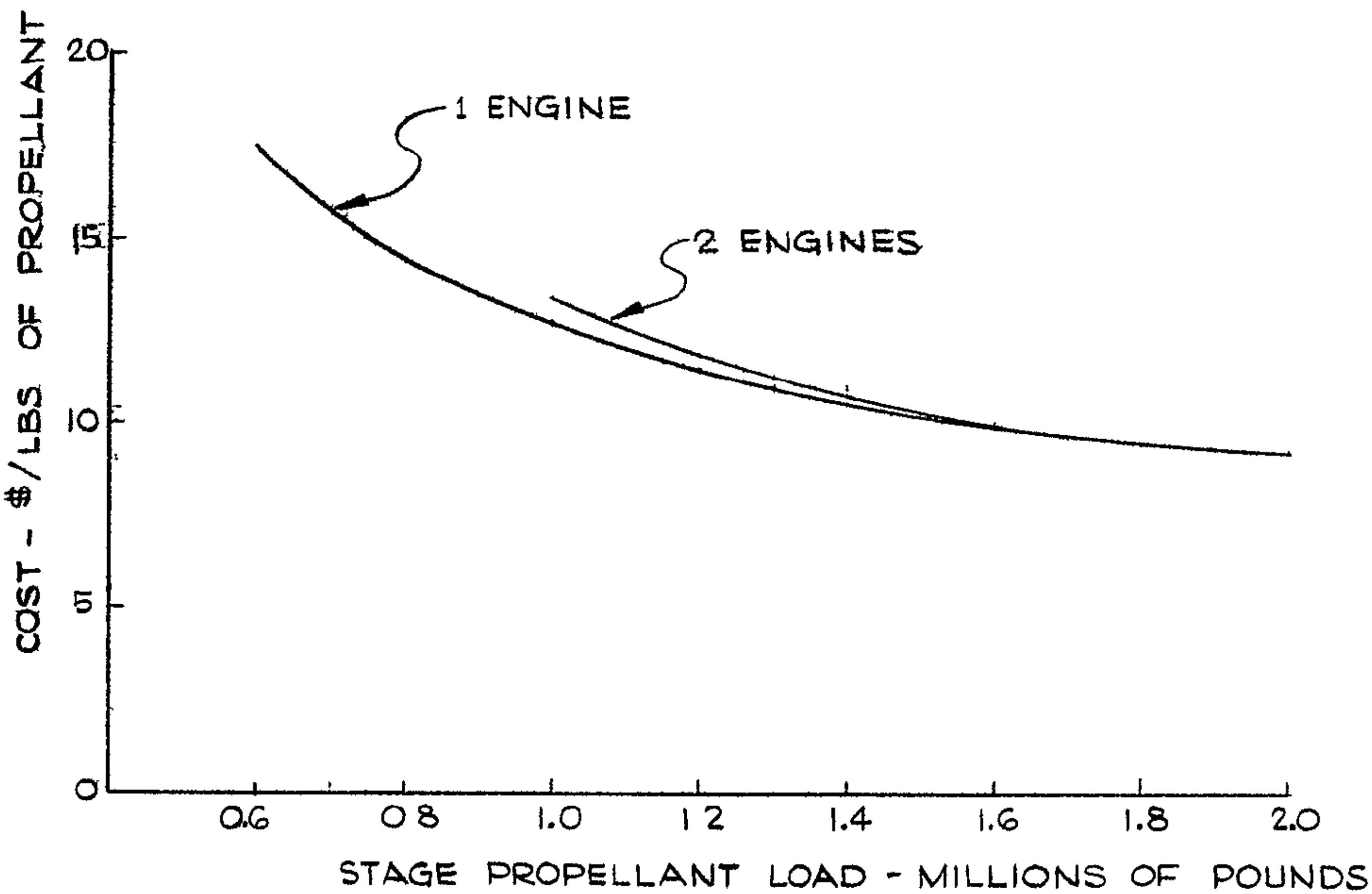


Figure 4-37

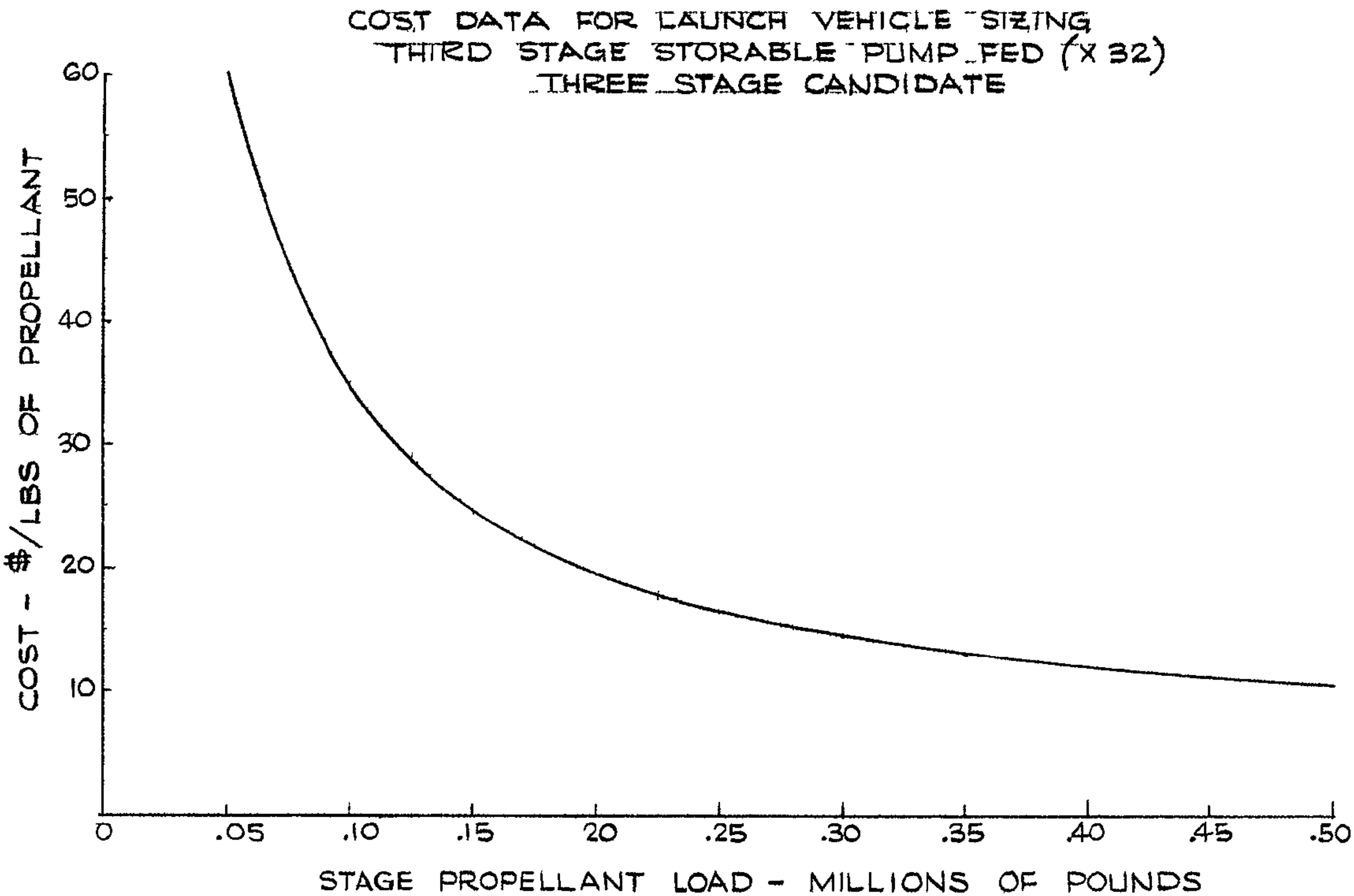

COST DATA FOR LAUNCH VEHICLE SIZING
THIRD STAGE STORABLE PUMP FED (X 32)
THREE STAGE CANDIDATE
COST - $/LBS OF PROPELLANT
60
50
40
30
20
10
0
.05
.10
.15
.20
.25
.30
.35
.40
.45
.50
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-38

COST DATA FOR LAUNCH VEHICLE SIZING
FIRST - SECOND INTERSTAGE STORABLE PUMP FED (X32)
THREE STAGE CANDIDATE

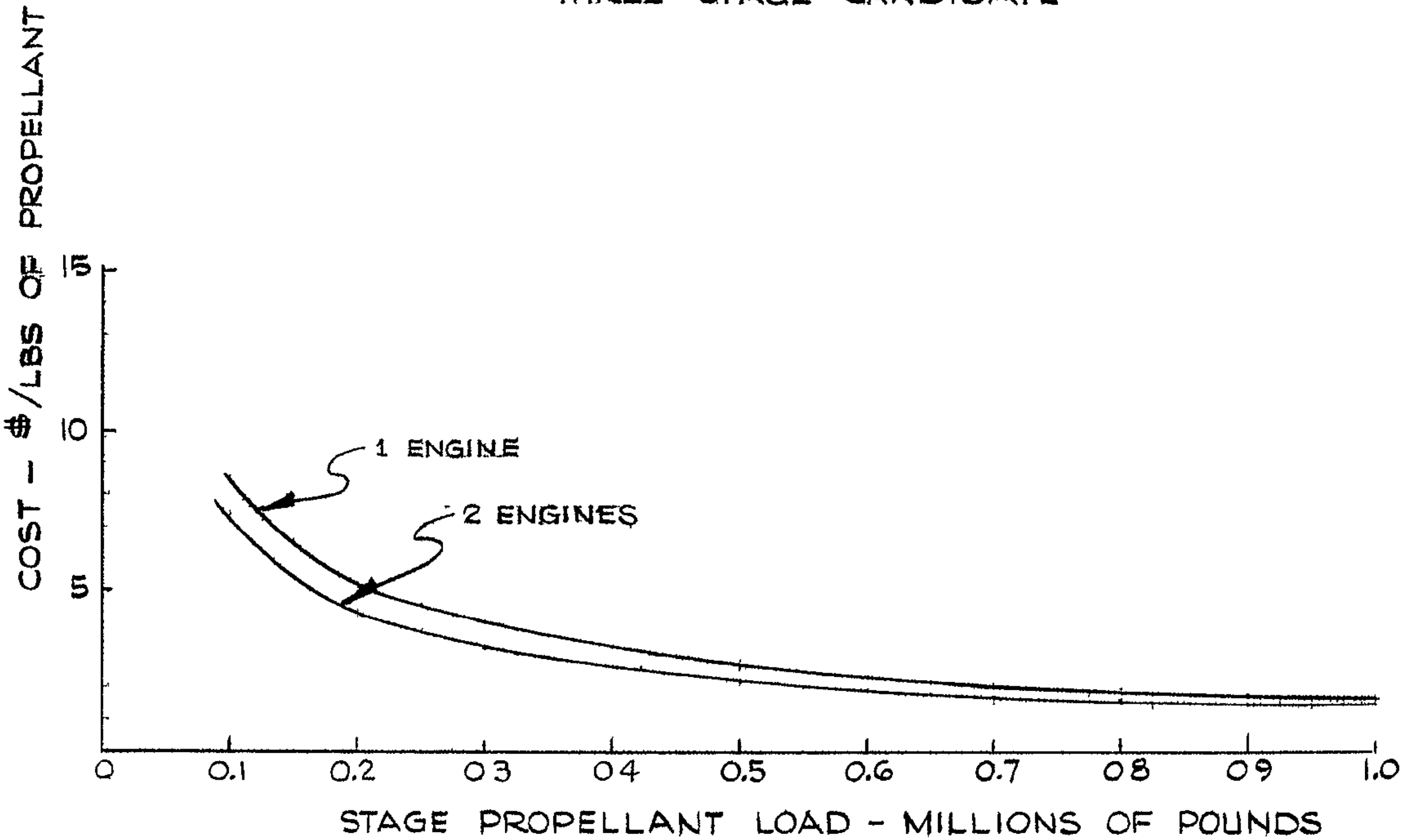


Figure 4-39

COST DATA FOR LAUNCH VEHICLE SIZING
SECOND STAGE STORABLE PUMP FED (X 32)
THREE STAGE CANDIDATE

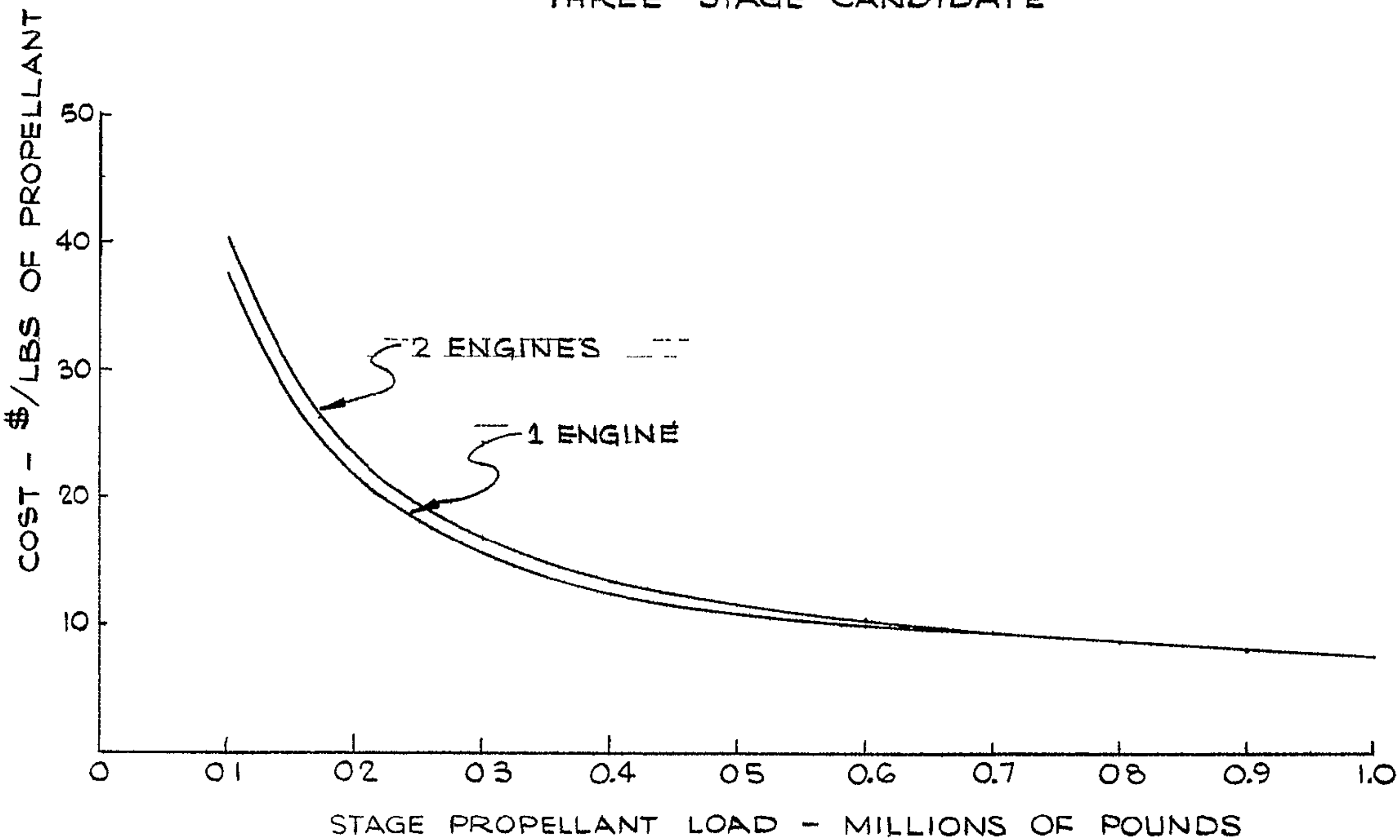


Figure 4-40

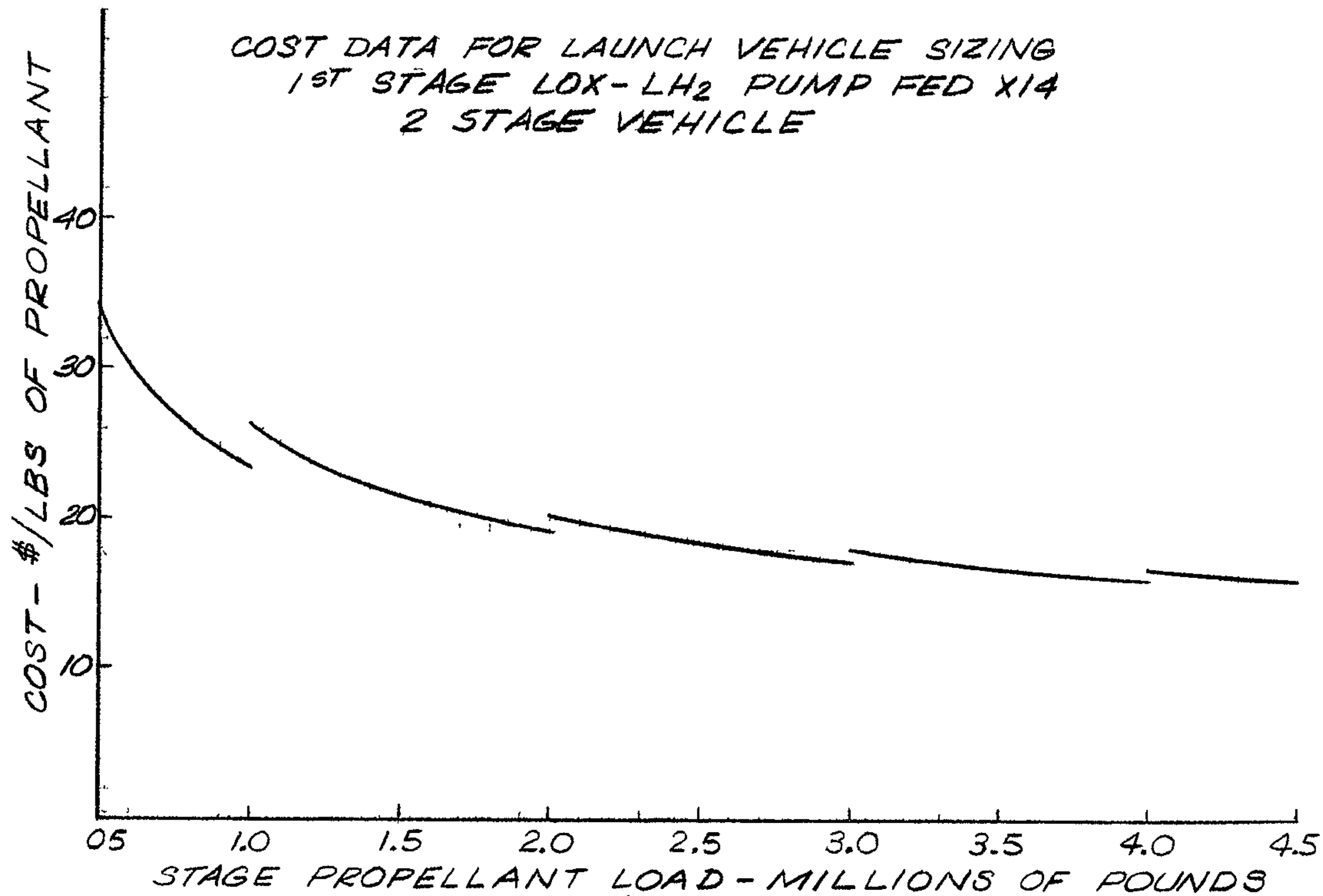

COST DATA FOR LAUNCH VEHICLE SIZING
1ST STAGE LOX-LH2 PUMP FED X14
2 STAGE VEHICLE
COST - $/LBS OF PROPELLANT
40
30
20
10
0.5
1.0
1.5
2.0
2.5
3.0
3.5
4.0
4.5
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-41

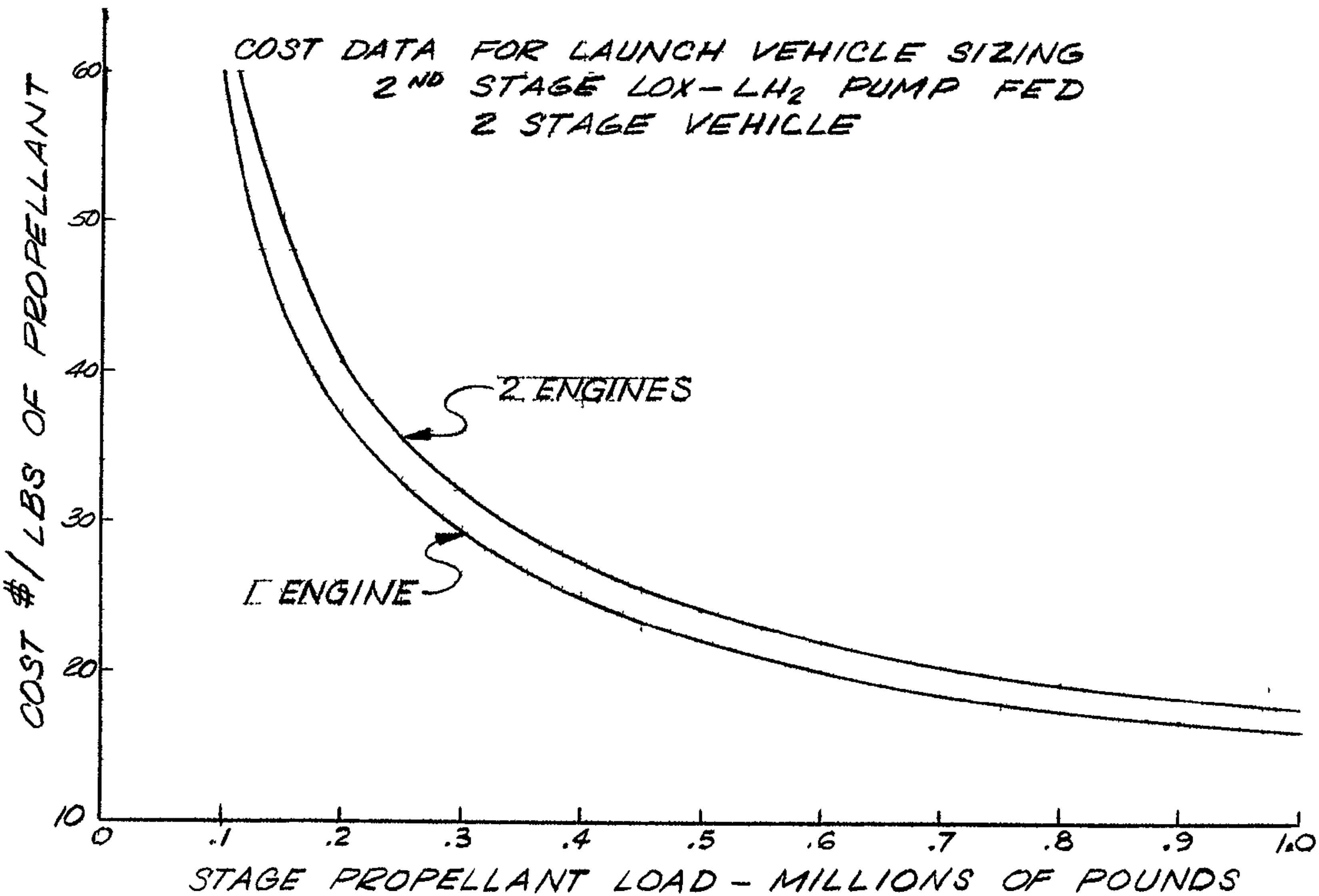

COST DATA FOR LAUNCH VEHICLE SIZING
2ND STAGE LOX-LH2 PUMP FED
2 STAGE VEHICLE
2 ENGINES
1 ENGINE
COST $/ LBS OF PROPELLANT
10
20
30
40
50
60
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-42

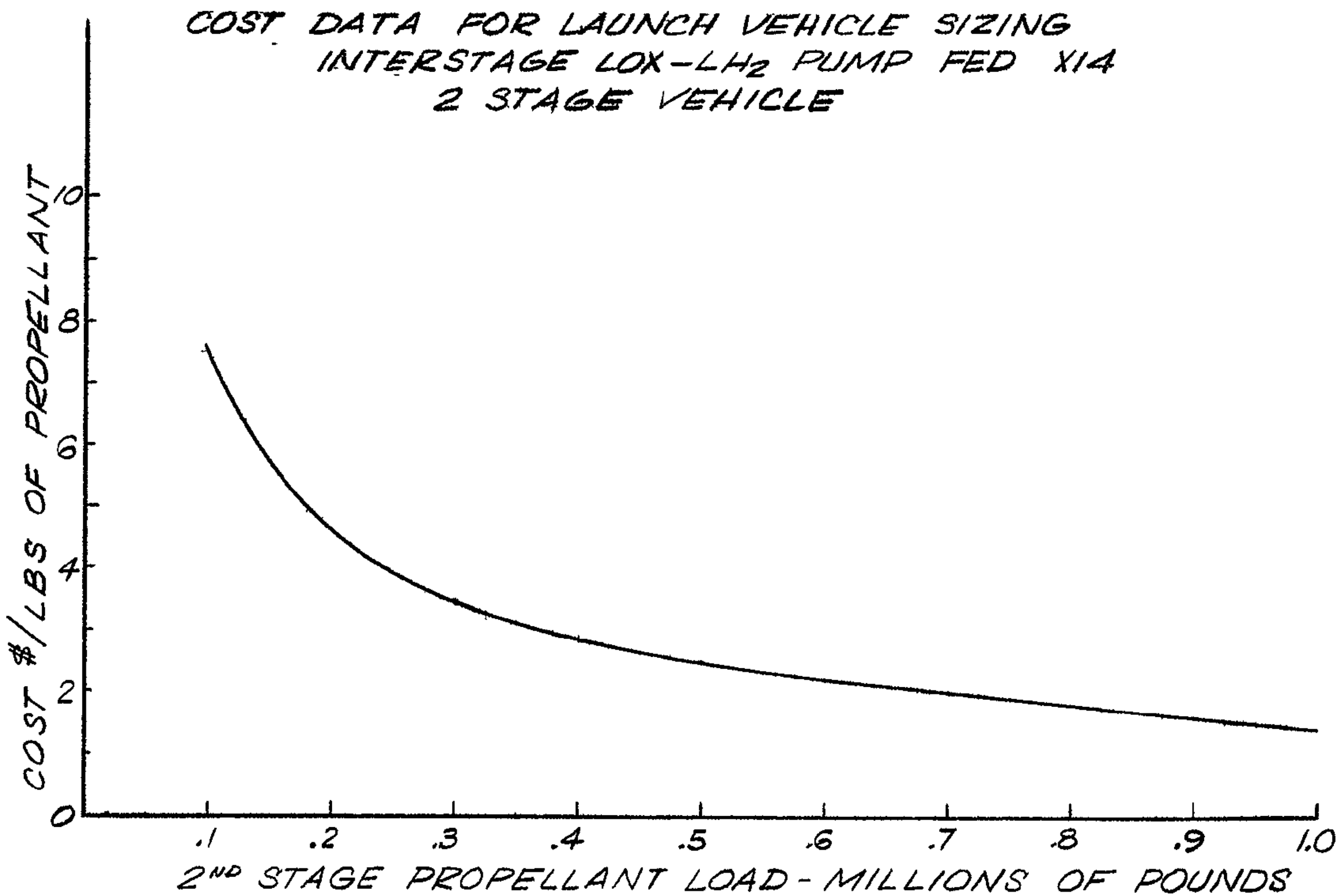

COST DATA FOR LAUNCH VEHICLE SIZING
INTERSTAGE LOX-LH2 PUMP FED X14
2 STAGE VEHICLE
COST $/LBS OF PROPELLANT
0
2
4
6
8
10
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
2ND STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-43

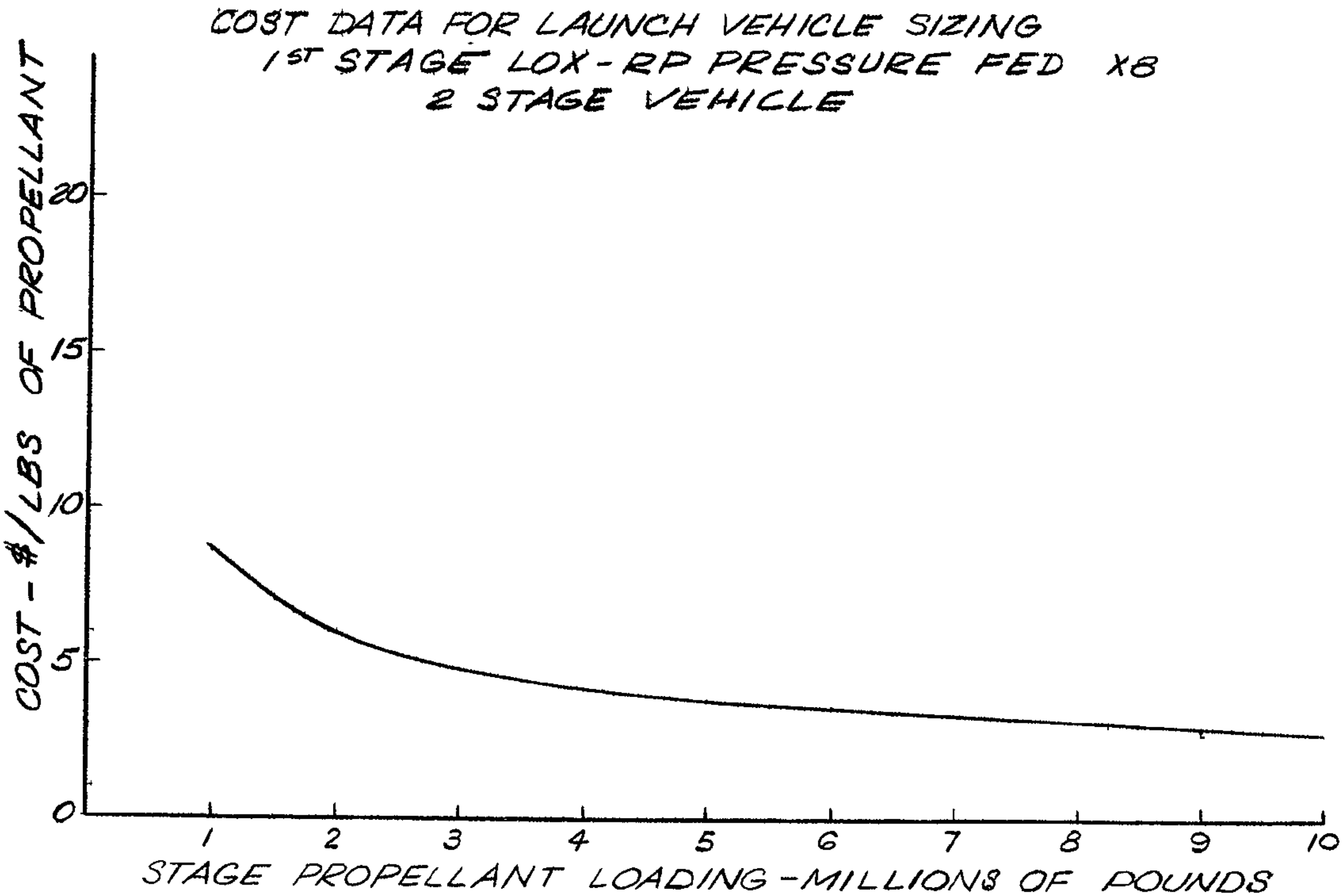

COST DATA FOR LAUNCH VEHICLE SIZING
1ST STAGE LOX - RP PRESSURE FED X8
2 STAGE VEHICLE
COST - $/LBS OF PROPELLANT
0
5
10
15
20
1
2
3
4
5
6
7
8
9
10
STAGE PROPELLANT LOADING - MILLIONS OF POUNDS

Figure 4-44

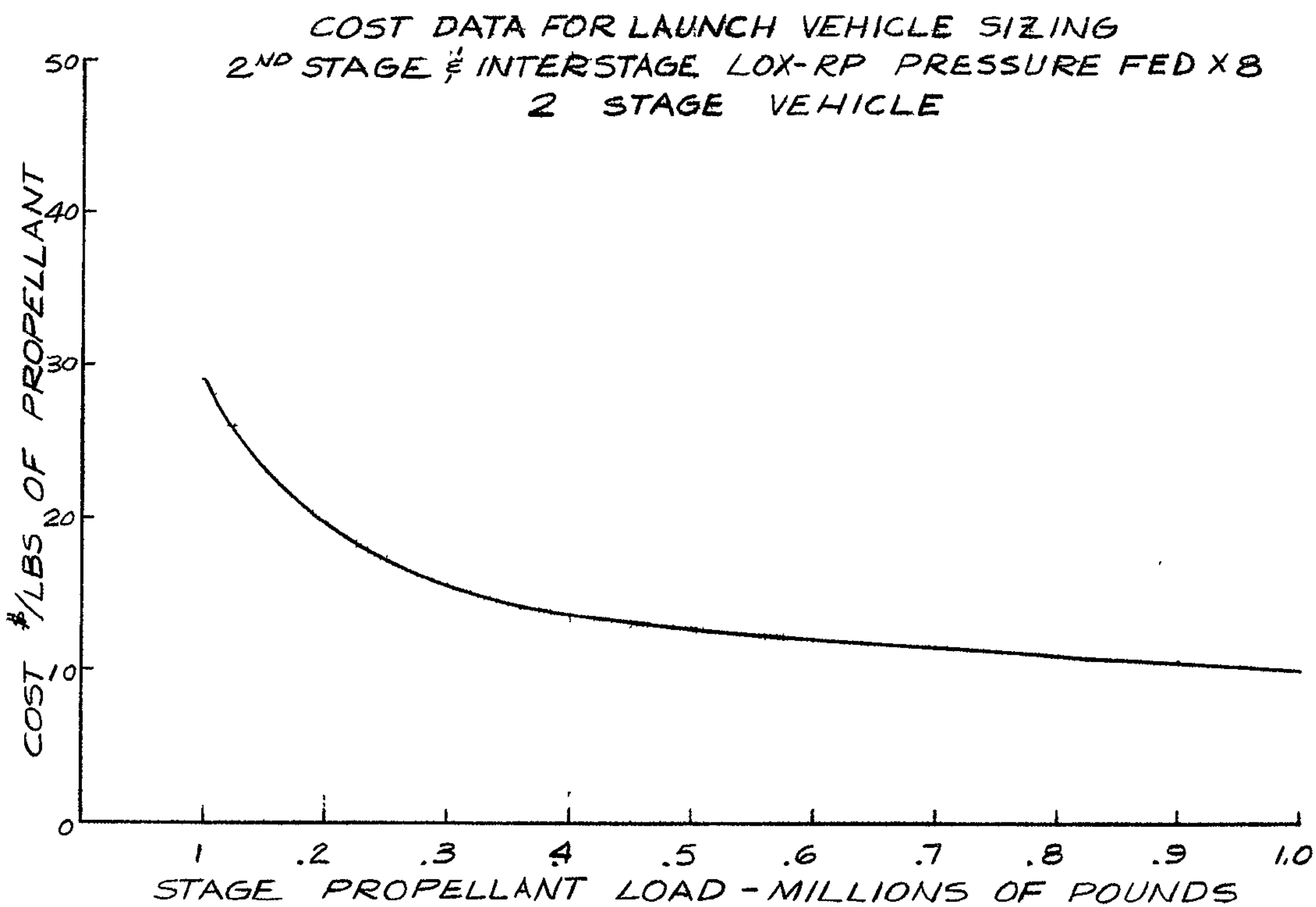

COST DATA FOR LAUNCH VEHICLE SIZING
2ND STAGE & INTERSTAGE LOX-RP PRESSURE FED X 8
2 STAGE VEHICLE
COST $/LBS OF PROPELLANT
50
40
30
20
10
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-45

COST DATA FOR LAUNCH VEHICLE SIZING X33
1ST STAGE SOLID 260
2 STAGE VEHICLE

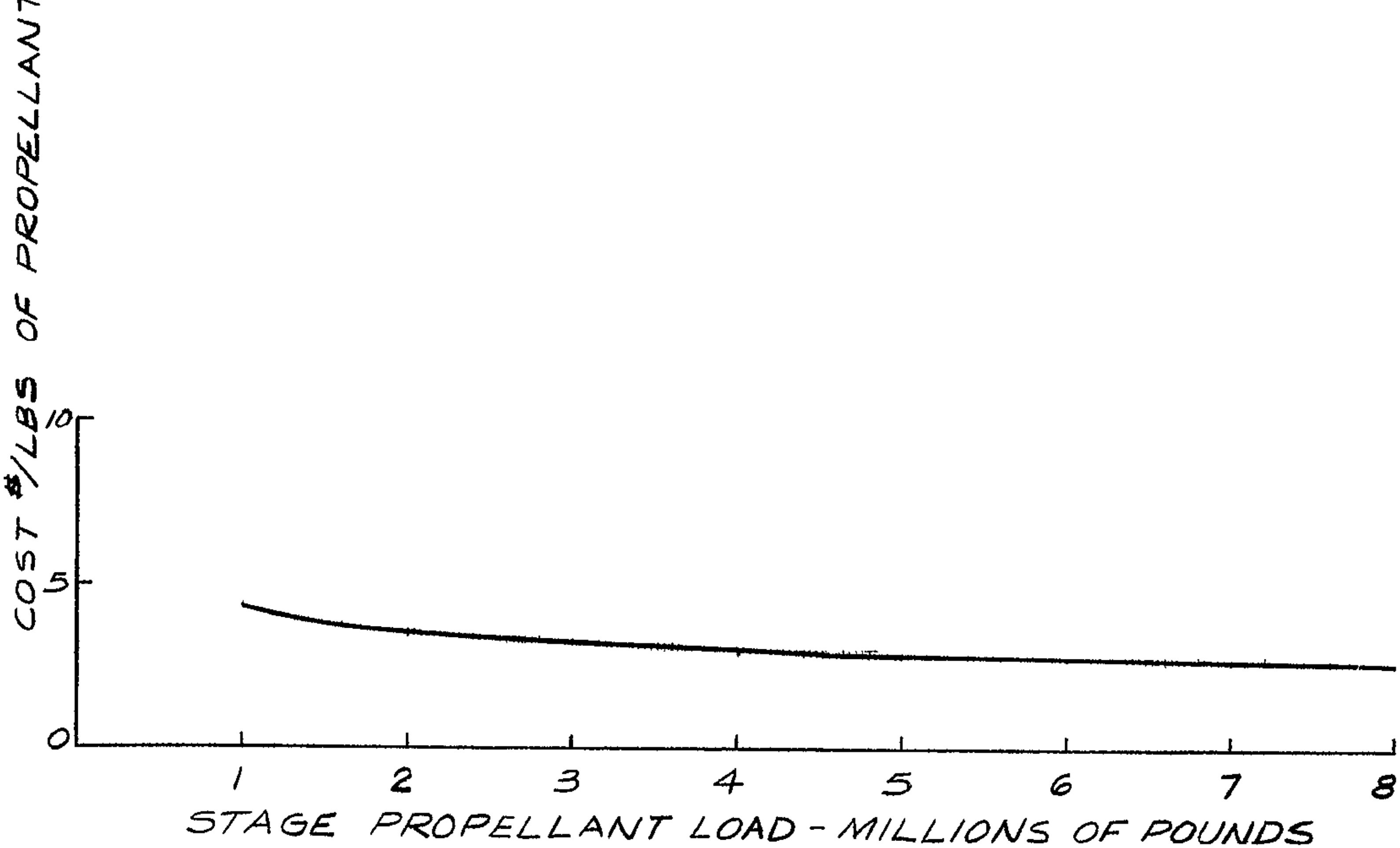


Figure 4-46

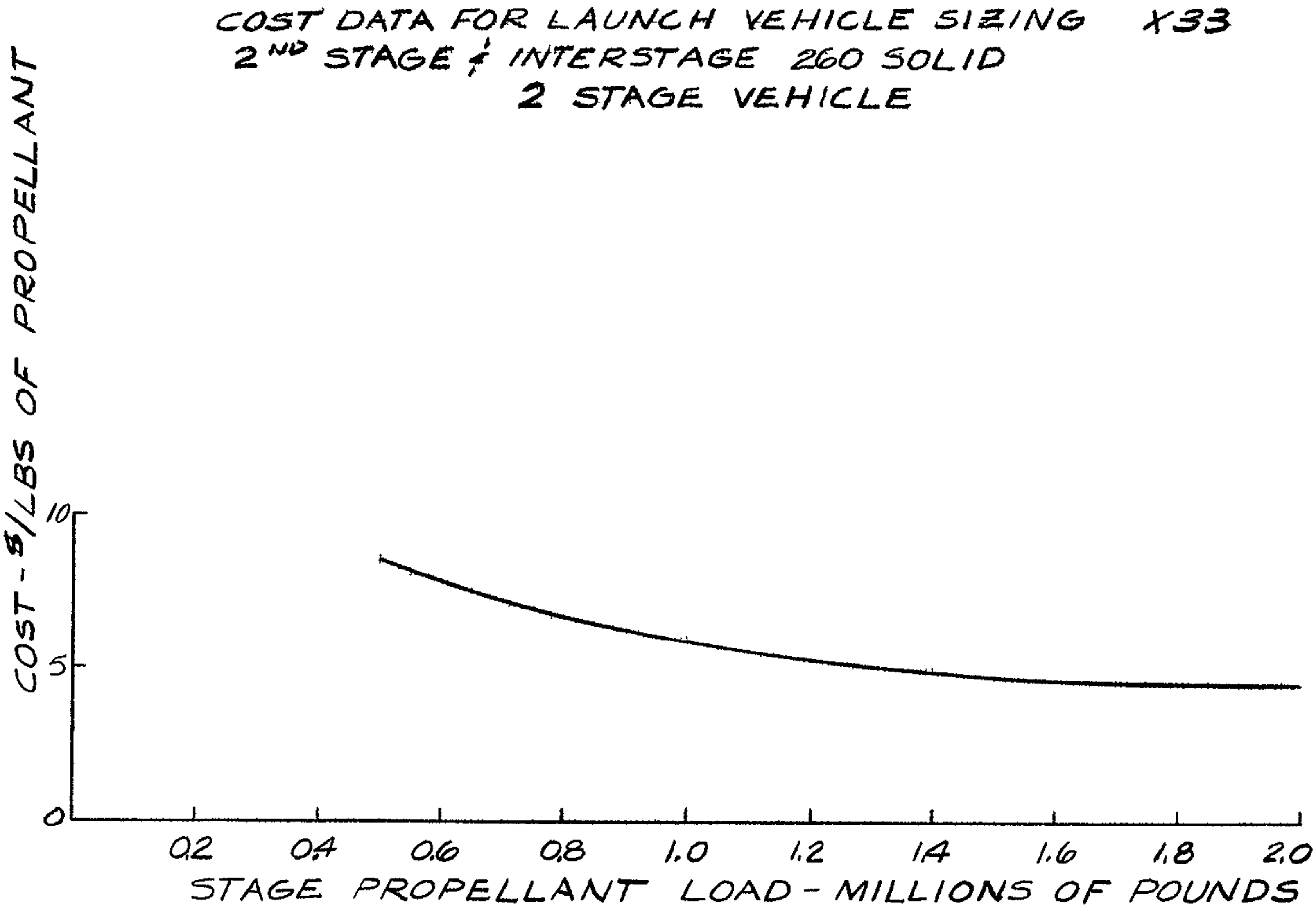

COST DATA FOR LAUNCH VEHICLE SIZING X33
2ND STAGE & INTERSTAGE 260 SOLID
2 STAGE VEHICLE
COST - $/LBS OF PROPELLANT
0
5
10
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
STAGE PROPELLANT LOAD - MILLIONS OF POUNDS

Figure 4-47

V COST SIZING AND LAUNCH OPERATIONS ANALYSES

This section describes the analyses conducted to define the minimum cost configuration of each of the candidate concepts and the related launch operations manpower requirements contributing to total system recurring cost

5 1 Cost Sizing Analysis - The objective of this phase of the study was to optimize the velocity distribution of each candidate booster using first unit minimum recurring cost as a criterion. All candidates have been cost sized for a baseline thrown weight of 82,500 lb (25,000-lb cargo) and velocity commensurate with a 100-nmi, 55° orbit. The results of this analysis are presented in Figures 5-1 through 5-5 In addition, the cost sensitivity of the more promising candidates to thrown weight, has been investigated This sensitivity is illustrated in Figure 5-6.

Cost sizing, as employed in this study, is defined as the determination of the propellant load of each stage of each launch vehicle concept that will result in the minimum recurring cost for a given mission. The approach taken to realize this objective was to first optimize each candidate for a minimum-growth factor or weight and then, utilizing the parametric weight and cost data described in Sections 4. 1 and 4. 2, systematically perturb the minimum growth factor configuration to determine an equivalent minimum-cost configuration To facilitate this analysis, an existing MDC computer program has been modified A flow chart of this program is presented in Table 5-1

Table 5-2 describes the mission that all candidates were sized to perform, also indicated are the thrust-to-weight ratios assumed for this analysis These ratios, 1 25 for first stages and 1.0 for upper stages, were selected as being representative of near optimum values, and assumed applicable to all candidates This parameter was used to determine the velocity losses, which have been represented by empirical relationships with thrust to weight as a major independent variable

The required mission velocity, 25,822 fps, resulted from the boost and orbital requirements shown in Table 5-3. The required orbital inclination of 55° in conjunction with a launch from Kennedy Space Center necessitated a 2 5° dog-leg maneuver This additional dog-leg increased the required mission velocity, 1010 fps. The remainder of the boost and orbital conditions bring the total required mission velocity to 25,822 fps. The total impulse velocity built into

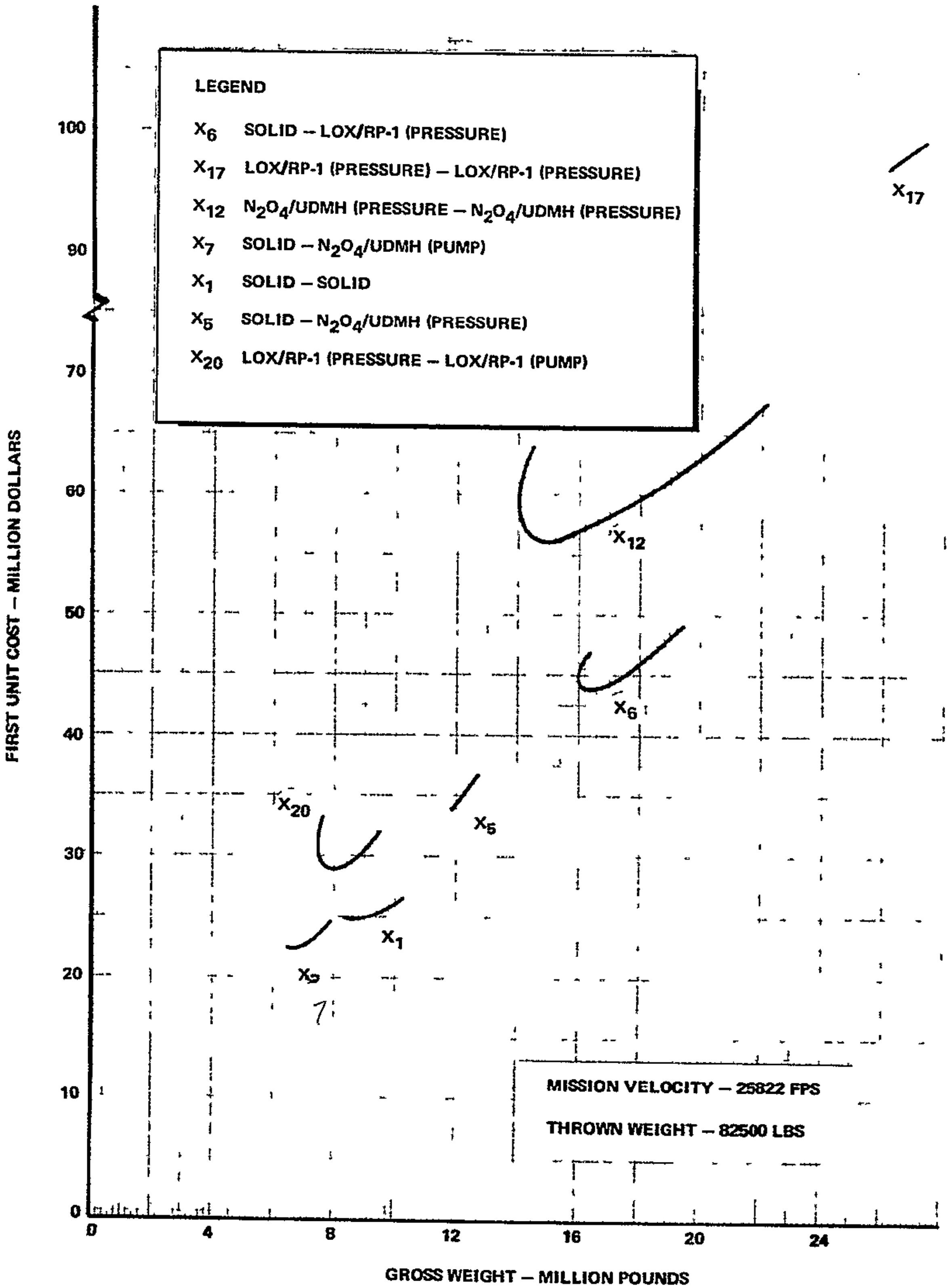


Figure 5-1. Launch Vehicle Sizing-Cost Variation for Two Stage Candidates

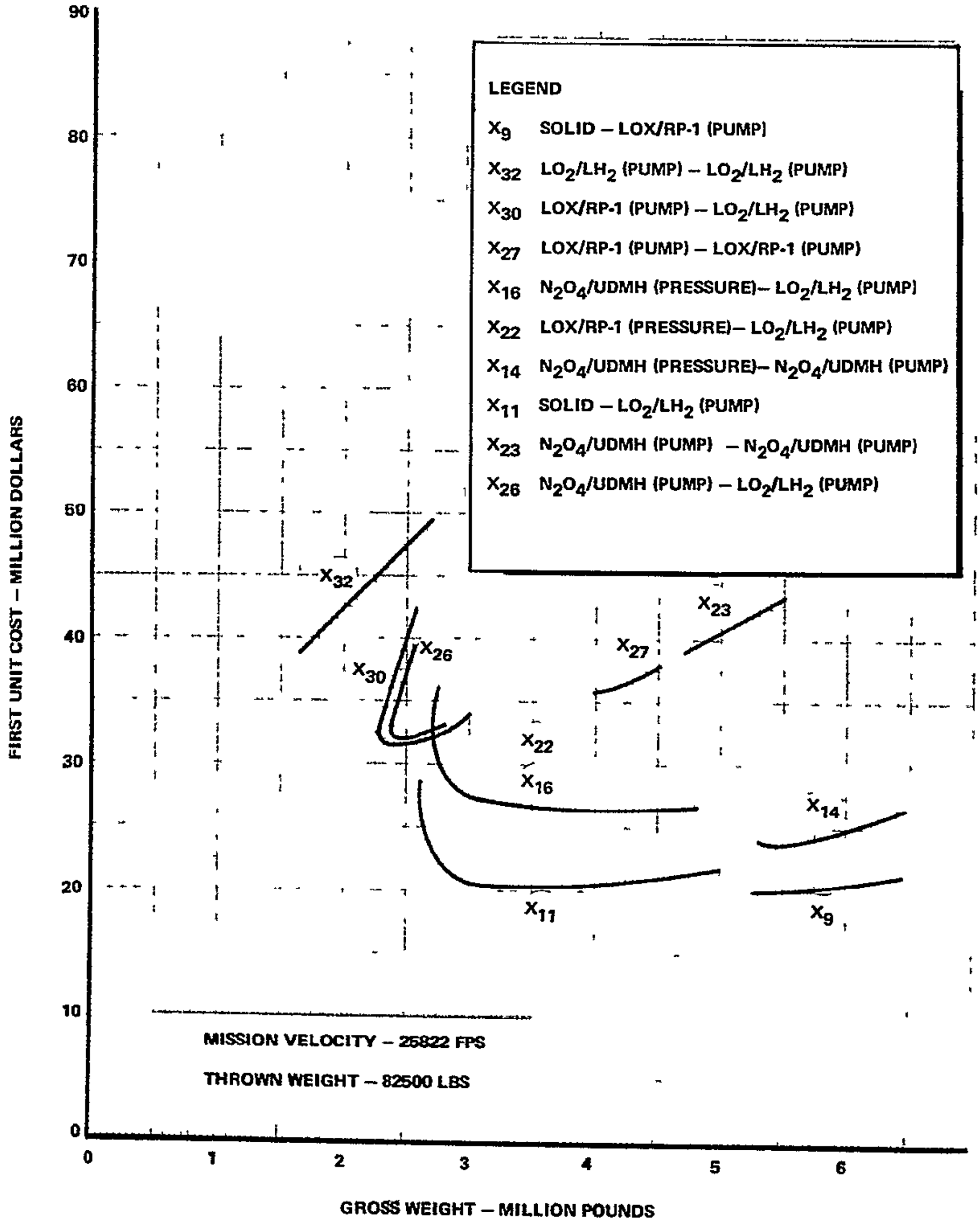


Figure 5-2 Launch Vehicle Sizing-Cost Variation for Two Stage Candidates

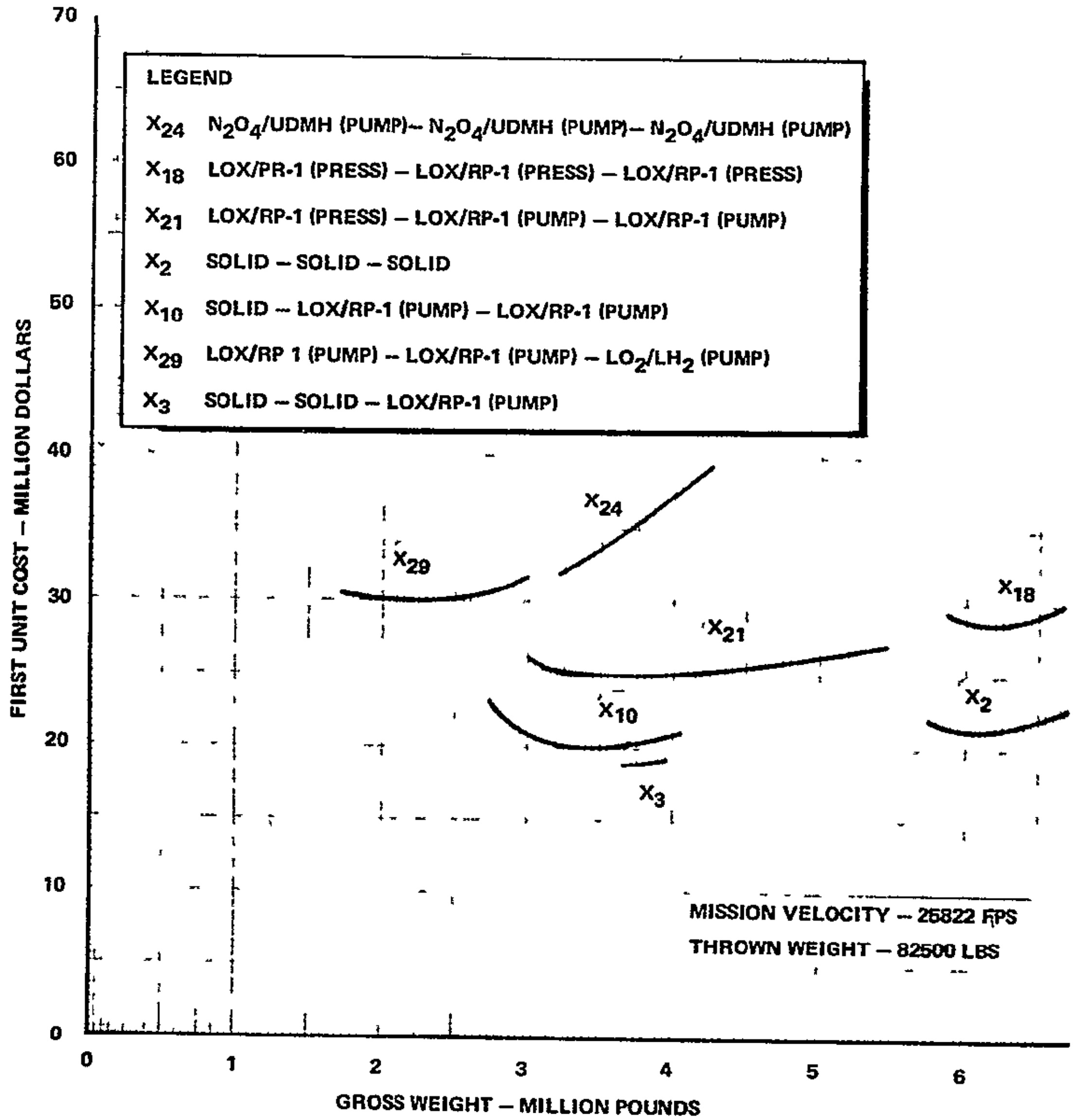


Figure 5-3. Launch Vehicle Sizing-Cost Variation for Three Stage Candidates

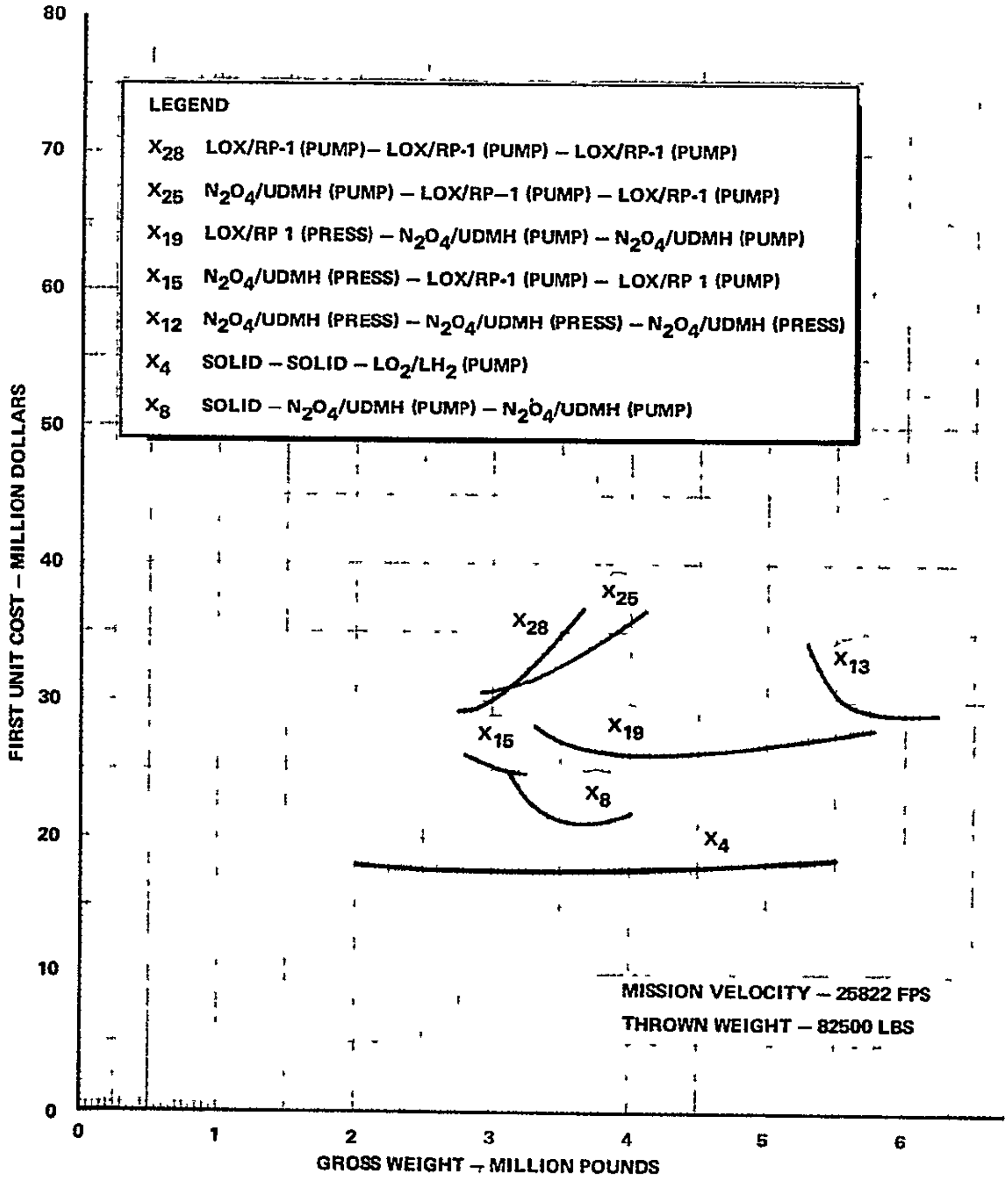


Figure 5-4 Launch Vehicle Sizing-Cost Variation for Three Stage Candidates

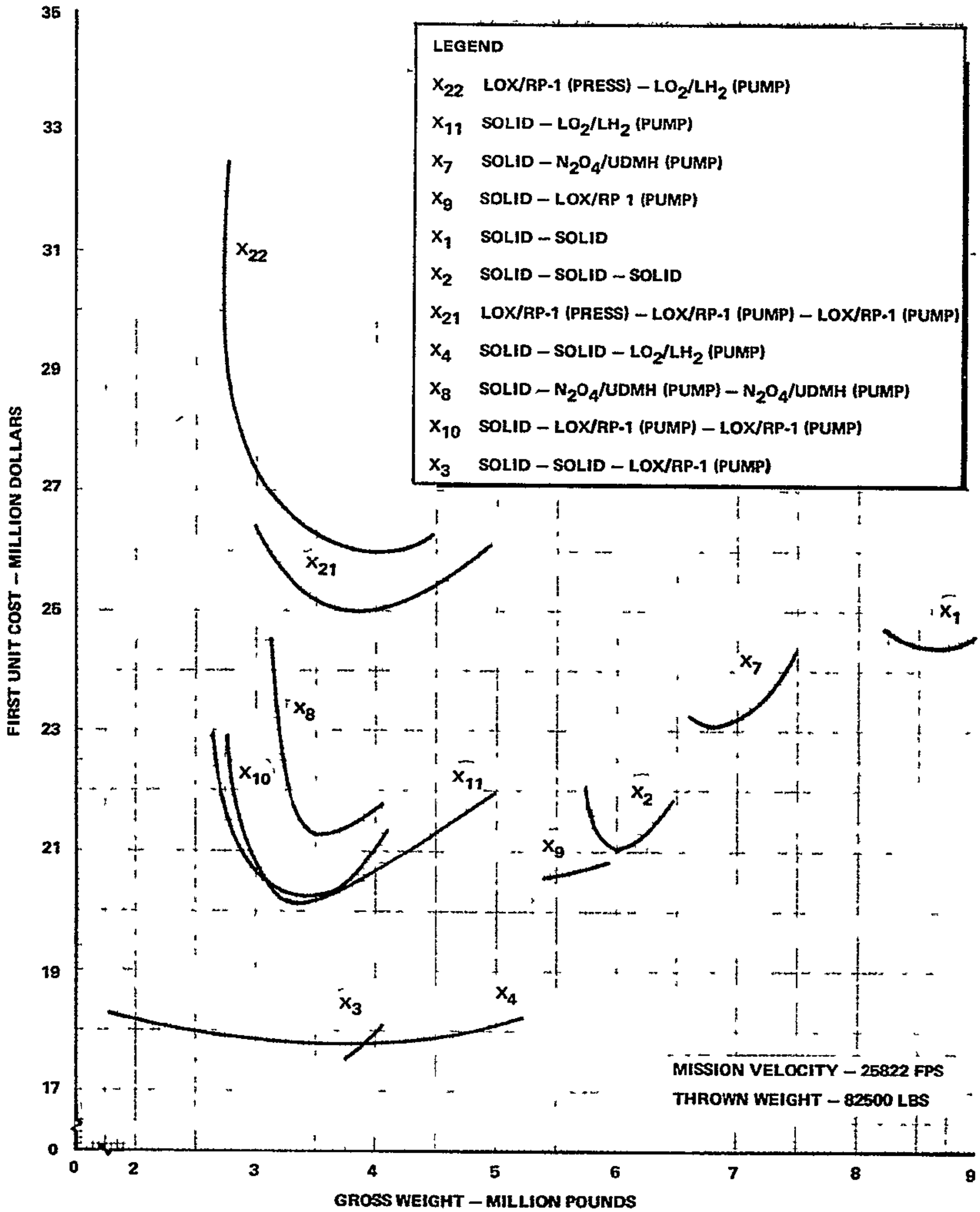


Figure 5-5. Launch Vehicle Sizing-Minimum Cost Candidates

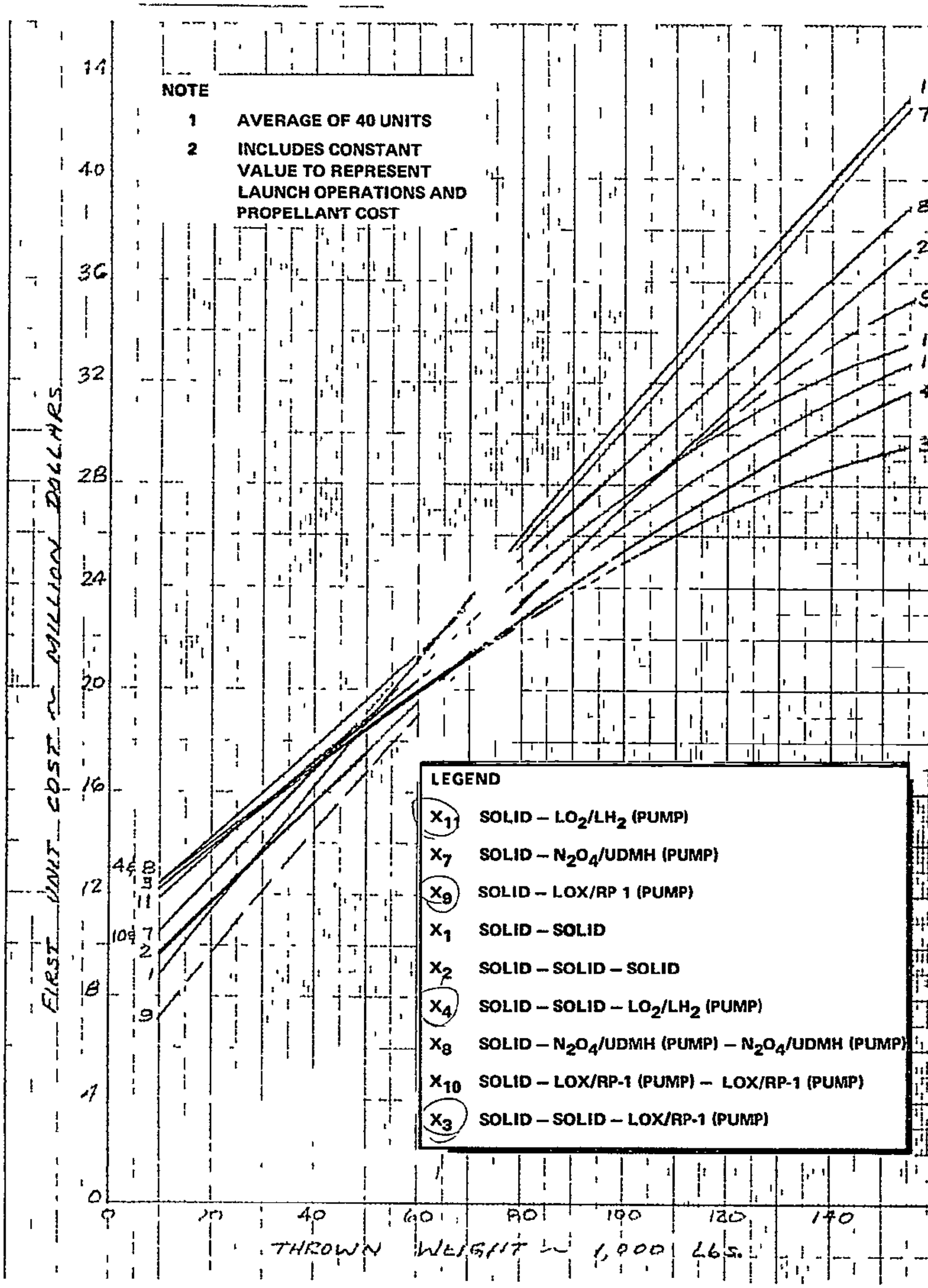


Figure 5-6 Cost Variation With Thrown Weight

Table 5-1

LAUNCH VEHICLE SIZING PROGRAM FLOW CHART

Input Data	→	Optimization	→	Output Data
$\lambda' = f(Wp)$		μ_1		$\$ = f(Wo)$
$\$ = f(Wp)$		μ_2		$\$ = f(W_{PAY})$
$Isp = f(h)$		μ_3		Vehicle Characteristics
T/W		Δ V Losses		Wp
Drag		Cost		Wo
Mission				W_{BO}
Payload				$W_{Structure}$
				ΔV's
				Etc

each vehicle is the sum of the mission velocity plus the velocity losses The velocity elements comprising the total impulsive velocity are delineated in Table 5-4

The data shown in Table 5-4 are for a two-stage launch vehicle (X_{30}) having a pump-fed LO_2/RP-1 first stage with a pump-fed LO_2/LH_2 second stage These data indicate the characteristics of both the minimum-weight and minimum-cost expendable launch vehicle As can be seen, in order to satisfy a mission velocity of 25,822 fps some 5,600 to 6,000 fps velocity capability had to be added to account for the velocity losses. In addition, Table 5-4 shows what is generally true for vehicles with dissimilar stage characteristics, that is, the minimum-cost

Table 5-2

EXPENDABLE LAUNCH VEHICLE MISSION

Orbital Altitude	=	100 nmi
Required Velocity	=	25,822 fps
$C_{D_{MAX.}} A/Wo$	=	0.0003 ft^2/lb
Payload Weight	=	82,500 lb
$\gamma_{B.O.}$	=	0^o
Launch (T/W)	=	1 25
Upper Stages (T/W)	=	1 0

Table 5-3

TOTAL REQUIRED MISSION VELOCITY

Orbital Altitude	=	100 nmi
Burnout Altitude	=	70 nmi
Perigee Velocity	=	25,690 fps
Orbital Inclination	=	55^{o}
2 5^{o} Dog-leg Maneuver	=	1010 fps
Earth Rotation	=	-930 fps
Launch Azimuth	=	44^{o}
Orbital Injection	=	52 fps
Total Required Velocity	=	25,822 fps

launch vehicle is larger than the minimum-weight vehicle This is because the least expensive stage assumes a greater portion of the mission velocity, thereby forcing the vehicle to become heavier but cost less. However, for a launch vehicle with all stages having similar weight and cost characteristics, since no one stage has a cost advantage over another, the minimum-growth factor configuration is also the minimum-cost configuration The results of the cost sizing analysis of all candidates considered are presented in Figures 5-1 through 5-3.

Figures 5-1 and 5-2 show the variations of launch vehicle first unit recurring cost with vehicle gross weight for two stage candidates. All candidate launch vehicles were sized for a constant payload weight of 82,500 lb. Inspecting both Figures 5-1 and 5-2, the following two-stage expendable launch vehicles appear to be potential minimum-cost candidates

X_9 - solid-LO_2/RP-1 (pump)

X_{11} - solid-LO_2/LH_2 (pump)

X_{22} - LO_2/RP-1 (pressure) - LO_2/LH_2 (pump)

X_7 - solid - N_2O_4/UDMH (pump)

X_1 - solid - solid

Although all the above launch vehicles have first unit cost between $20 million to $25 million dollars, their gross weights at launch vary from 3 to 10 million lb. The heaviest vehicle of the above minimum cost two-stage candidates is the all solid configuration (X_1), the lightest is a solid first stage with a pump-fed LH_2

Table 5-4

TYPICAL (X_{30}) COST SIZING DATA

Items	Minimum-Weight Vehicle		Minimum-Cost Vehicle
Growth Factor	27 86		29 40
Mass Ratios μ_I	2 46		3 61
μ_{II}	5 26		4.16
Impulse Velocity ΔV_1	31,454	fps	31,887
ΔV_1 I	8,646	fps	12,301
ΔV_1 II	22,808	fps	19,586
Gravity Losses	4,837	fps	5,378
Back-Pressure Losses	326	fps	326
Drag Losses	360	fps	360
Gross Weight	2 298	million	2 425
Cost	$32.63	million	31.64

second stage. Using recurring costs as a criterion, this difference in weight is not significant However, if nonrecurring launch cost were considered, the heavier vehicles would be much less attractive Another interesting characteristic of these curves is an apparent cost insensitivity of candidates with LO_2/LH_2 second stages to reductions in second-stage propellant load below the minimum-cost configuration This characteristic makes feasible the substitution of existing hardware in place of the cost-optimized second stage The most significant result of this analysis, however, is the dominance of solid first-stage candidates in the low-cost category This is primarily due to their significantly lower cost.

Cost variations for candidate three-stage launch vehicles are shown in Figures 5-3 and 5-4 These data lead to the same conclusions as mentioned for the two-stage candidates. In addition, when comparing the three-stage with the two-stage candidates, the obvious advantages from staging is quite clear As an example, the two-stage solid (X_1) has a gross weight of 8 million lb, while a three-stage solid (X_2) performs the same mission for 6 million lb The following three-stage launch vehicles appear to be potential minimum cost candidates

X_2 - solid-solid-solid

X_{10} - solid - LO_2/RP-1 (pump) - LO_2/RP-1 (pump)

X_{21} - LO_2/RP-1 (pressure) - LO_2/RP-1 (pump) - LO_2/RP-1 (pump)

X_8 - solid - N_2O_4/UDMH (pump) - N_2O_4/UDMH (pump)

O_3 - solid - solid - LO_2/RP-1 (pump)

O_4 - solid - solid - LH_2/LH_2 (pump)

The above three-stage minimum cost candidates have costs which range around $20 million As has been mentioned, the cost data shown in Figures 5-1 to 5-5 are first-unit recurring hardware costs, with the exception that the solid-motor candidates include propellant. Therefore, the other launch vehicles will increase in cost when the cost of propellant is considered The costs of launch operations and propellant have been estimated and are shown in the following sections

5 2 Launch Operations Analysis - To provide a basis for candidate concept screening, preliminary estimates were made of launch operations manpower requirements which contribute to system recurring costs These estimates, covering the range of operations from stage receipt at the launch site through vehicle build-up, checkout and launch, were based upon the following assumed ground rules

(1) Each stage or major element of the system has undergone a complete simulated system checkout as part of acceptance at the factory prior to delivery to the launch site

(2) Each stage is complete in its configuration at the time of receipt at the launch site, i e , no shortages exist and no modifications, or mission peculiar kit installations are to be made

(3) Normal launch operations were assumed to be conducted on a 1-shift, 8 hour workday, 5 days per week, 250 work days per year basis up to the time of starting a final flight readiness test from this point on a 2-shift, 6 day work week was assumed, except for actual countdown, or simulated countdown activities which were assumed to be conducted on a 24-hour day basis

(4) A single launch vehicle contractor was assumed responsible for the complete launch vehicle operations (receipt through launch) regardless of the type or numbers of stages involved

(5) Only launch vehicle personnel requirements dictated by candidate vehicle configuration, size, and complexity were to be estimated Estimates did not include spacecraft or payload operations personnel, customer support or other launch site external support personnel, i e , Range Safety, fire protection, medical, and other services

(6) Minimum off-pad stage receiving-inspection activities were assumed, major launch preparation activities, including launch vehicle assembly and checkout, were to be accomplished at the launch pad

Following these assumed ground rules, the approach taken was first, to define for a typical baseline vehicle a preliminary estimate of launch operations manpower requirements, and subsequently to assess the effects of multi-staging, variation in types of propellant systems (solid vs liquid, cryogenic vs storable, pump fed vs pressure fed, etc) for each of the candidate vehicles selected in the cost sizing analysis

As a point of departure for the analysis, data generated under Air Force Contract F33615-69-C-1327, Advance Configuration Concepts Study, were utilized to define the baseline vehicle requirements These data, for a launch vehicle consisting of a 260 in solid motor first stage and a lower cost version of the current Saturn S-IVB liquid oxygen-liquid hydrogen second stage, were in turn derived from the results of an in-depth study of the operational manpower requirements for a 260 in solid boosted S-IVB vehicle conducted under NASA Contract NAS 10-4802, Launch Facilities and Operations for Large Solid Motors Study The lower cost version of the S-IVB, currently undergoing in-house study at MDAC, offers significant reductions in operational manpower requirements and checkout time as a result of overall system simplification and critical subsystem elimination The manpower requirements for these previously studied vehicles were considered to be typical for a baseline vehicle consisting of a solid motor first stage and liquid oxygen-liquid hydrogen, pump fed second stage (Candidate XII)

Basic operational manpower requirements thus derived were then compared with actual manpower statistics for a typical operational program (Thor-Delta) to establish the realism of the estimates, and to identify appropriate, realistic factors for support personnel and other miscellaneous manpower requirements

Table 5-5 summarizes the manpower requirements for the candidate vehicles which were subsequently utilized to estimate the launch operations recurring costs reflected in Section VI (Table 6-1)

Table 5-5 (page 1 of 3)

MANPOWER REQUIREMENTS ESTIMATES. LAUNCH OPERATIONS

Candidate	Configuration	Operations	Quality Control	Engineering	Support	Total
03	Solid, Solid, Pump LO_2/RP-1	184	52	92	81	409
04	Solid, Solid, Pump LO_2/LH_2	184	52	92	81	409
X11	Solid, Pump LO_2/LH_2	161	45	79	71	356
X9	Solid, Pump LO_2/RP-1	160	44	78	70	352
X10	Solid, Pump LO_2/RP-1, Pump LO_2/RP-1	205	58	102	91	456
X2	Solid, Solid, Solid	129	35	62	56	282
X8	Solid, Pump Storable, Pump Storable	200	56	100	89	445
X7	Solid, Pump Storable	145	40	72	64	321
X1	Solid, Solid	102	27	48	44	221
X22	Press. RP, Pump LO_2/LH_2	164	45	80	72	361
X21	Press RP, Pump RP Pump RP	218	62	109	96	485
X14	Press Storable, Pump Storable	150	42	75	66	333
X15	Press. Storable, Pump RP, Pump RP	215	60	107	95	477

Table 5-5 (page 2 of 3)

Candidate	Configuration	Operations	Quality Control	Engineering	Support	Total
X16	Press Storable, Pump LO_2/LH_2	168	46	83	74	371
X19	Press RP, Pump Storable, Pump Storable	205	58	123	91	477
X20	Press RP, Pump RP	164	45	80	72	361
X28	Pump RP, Pump RP, Pump RP	201	56	100	88	445
X30	Pump RP, Pump LO_2/LH_2	175	48	86	77	386
X5	Solid, Press Storable	142	39	70	62	313
X18	Press RP, Press RP, Press RP	188	52	93	83	416
X13	Press Storable, Press. Storable, Press Storable	174	48	87	76	385
X26	Pump Stor., Pump LO_2/LH_2	170	47	84	75	376
X25	Pump Stor., Pump RP, Pump RP	217	61	104	96	478
X27	Pump RP, Pump RP	145	40	70	64	319
X24	Pump Stor., Pump Stor , Pump Stor.	184	51	91	81	407

Table 5-5 (page 3 of 3)

Candidate	Configuration	Operations	Quality Control	Engineering	Support	Total
X32	Pump LO_2/LH_2, Pump LO_2/LH_2	161	44	78	71	354
X23	Pump Storable, Pump Storable	134	36	65	58	293
X6	Solid, Pressure RP	145	40	72	64	321
X12	Press. Storable, Press Storable	130	35	62	56	283
X17	Press RP, Press Storable	137	36	65	60	298

VI LAUNCH VEHICLE CONCEPT SELECTION AND CONCLUSIONS

As a result of this study, it has been possible to select five low-cost launch vehicle concepts that deserve further analysis. Table 6-1 contains a listing of all 32 candidates in the order of increasing costs The lower cost concepts head the list

The first two columns of Table 6-1 represent the hardware cost (ready to ship to the launch site) of the first unit and the average of 40 units respectively The first column includes the propellant costs of the solids only, the second column however includes the cost of all propellants The third column represents the wages of the launch operations crew, and the fourth column represents a total cost, without launch site housekeeping support, of the average of 40 units This last column provided was the primary basis for the section of low cost candidates

Figure 5-6, which shows cost versus thrown weight to orbit, was also used as an aid in the selection of low cost candidates At the nominal thrown weight of 82,500 lb (25,000 lb of cargo) the three-stage solid ranked a close sixth However, as weight to orbit was increased to 150,000 lb, there was a substantial difference in its cost and the next lower cost candidate At lower thrown weights it looked quite competitive, but was no better than four of the other concepts Thus, for the purpose of this study the three-stage solid was not included as a candidate concept for further analysis.

The five candidates that were selected as the most likely low-cost expendable launch vehicle concepts are as follows

1. Solid first stage + solid second stage + LO_2/RP-1 third stage
2. Solid first stage + solid second stage + LO_2/LH_2 third stage
3. Solid first stage + LO_2/LH_2 second stage.
4. Solid first stage + LO_2/RP-1 second stage + LO_2/RP-1 third stage
5 Solid first stage + LO_2/RP-1 second stage

All liquid propellant stages, of the selected concepts are, pump-fed and all stages subscribed to the low cost philosophy described in Sections 3.1 and 4.2. The five lowest cost concepts utilize solid propellants in the lower stages Three of them have three stages and two have two stages. A cursory examination indicates that the three-stage approach may be the best if other factors are considered the three-stage vehicles are about 50% lighter than the two-stage vehicles, they will have lower maximum acceleration levels, and they are more efficient as the thrown weight increases.

Table 6-1 (page 1 of 3)

CANDIDATE RANKING BY RECURRING COST
($ MILLIONS)

Candidate	Configuration	1st Unit Hardware Cost	Average Hardware and Propellant Cost 40 Units	Launch OPS Cost	Average Launched Cost 40 Units
03	Solid, Solid, Pump LO_2/RP-1	17 6	12 1	4 0	16 1
04	Solid, Solid, Pump LO_2/LH_2	17 8	12.6	4 0	16 6
X11	Solid, Pump LO_2/LH_2	20 2	13.9	3 5	17 4
X9	Solid, Pump LO_2/RP-1	20 5	14.2	3 4	17 7
X10	Solid, Pump LO_2/RP-1, Pump LO_2/RP-1	20.3	13.7	4 5	18 2
X2	Solid, Solid, Solid	22 1	15.9	2 8	18 7
X8	Solid, Pump Storable, Pump Storable	22 5	15.4	4 4	19 8
X7	Solid, Pump Storable	23 3	16.7	3 2	19 9
X1	Solid, Solid	24 2	17 9	2.2	20 1
X22	Press. RP, Pump LO_2/LH_2	26 0	17 8	3 6	21.4
X21	Press. RP, Pump RP, Pump RP	25.0	16.6	4 8	21 4
X14	Press Storable, Pump Storable	23 6	18 4	3.3	21 7

Table 6-1 (page 2 of 3)

Candidate	Configuration	1st Unit Hardware Cost	Average Hardware and Propellant Cost 40 Units	Launch OPS Cost	Average Launched Cost 40 Units
X15	Press. Storable, Pump RP, Pump RP	24.9	17 3	4 7	22 0
X16	Press. Storable, Pump LO_2/LH_2	26.3	18 7	3 7	22 4
X19	Press RP, Pump Storable, Pump Storable	26.1	18.0	4 7	22 7
X20	Press. RP, Pump RP	29 2	19 2	3 6	22.8
X28	Pump RP, Pump RP, Pump RP	29.0	19 0	4 4	23.4
X30	Pump RP, Pump LO_2/LH_2	31 6	20 5	3 8	24 3
X5	Solid, Pressure Storable	33.6	21.7	3 1	24.8
X18	Press. RP, Pressure RP, Pressure RP	29.2	20.9	4 1	25 0
X13	Press Stor., Pressure Storable, Press Storable	28 4	21 3	3 8	25 1
X26	Pump Stor., Pump LO_2/LH_2	31 8	21 4	3.7	25 1
X25	Pump Stor , Pump RP, Pump RP	31.0	20.9	4 7	25 6

Table 6-1 (page 3 of 3)

Candidate	Configuration	1st Unit Hardware Cost	Average Hardware and Propellant Cost 40 Units	Launch OPS Cost	Average Launched Cost 40 Units
X27	Pump RP, Pump RP	35.6	22 7	3 2	25 9
X24	Pump Stor., Pump Storable, Pump Storable	34.3	24.4	4.0	28 4
X32	Pump LO_2/LH_2, Pump LO_2/LH_2	38.3	25.2	3 5	28 7
X23	Pump Storable, Pump Storable	39 0	26 8	2 9	29 7
X6	Solid, Pressure RP	43 9	27 3	3 2	30 5
X12	Press Stor., Pressure Storable	56.1	40.7	2 8	43 7
X17	Press. RP, Pressure Storable	98 8	65 6	2 9	68 5

Notes

1. Excludes liquid propellants.
2 Includes propellants

The pressure-fed liquids followed the solids as low-cost first-stage concepts, but as upper stages they were always substantially more expensive This study has shown that for low cost, launch vehicles with one or two solid-propellant lower stages offer a clear cut advantage over the other propulsion concepts It is also shown that launch vehicles with pump-fed LO_2/RP-1 upper stages appear to have lower cost than the other upper stages investigated

APPENDIX

I PRELIMINARY COST ANALYSIS (EXPENDABLE LAUNCH VEHICLES)

During the initial phase of the study, the costs of existing vehicles and systems were compared This effort consisted of analyzing existing, published costs figures rather than developing new data The purpose of this comparison was to isolate the factors that caused the costs of one program to be different than the costs of another program In order to facilitate recognition of real cost differences, the actual dollar figures were not used but all costs were reduced to percentages to permit comparing the relative expenditures for the various parts of one program with the relative expenditures for the same parts of the other programs. Each of the programs investigated have widely different total cost If the comparison between elements within a program had been made in terms of dollars, it would have been difficult to make a meaningful comparison between the programs For instance, no study is necessary to discover that more dollars are spent for Saturn operations than were spent for Thor-Delta operations However, it is important to know that a larger part of the total Thor-Delta program cost is spent for operations than is spent for operations in the Saturn programs This relationship would not be readily apparent and might not be recognized at all if actual dollar figures had been used to compare the programs

In addition, if actual dollar figures had been used, too much attention would be devoted to justifying the actual dollar figure quoted and not enough to the relationships between the cost elements. For purposes of this survey, the cardinal concern is to discover areas where costs should be investigated These areas are more easily identified by calling attention to the different cost relationships existing between programs and isolating the reasons for these differences rather than by calling attention to the fact that more dollars are spent on one program than are spent on another program Therefore the following pie charts compare the cost relationships in percentages of total cost rather than quoting actual dollar figures However the size of the pies have been varied in proportion to the dollar cost of each program

The information derived from the comparison will be used as a guide for decreasing costs in the program being studied. The resulting conclusions will be applied within the framework of decreasing costs without sacrificing safety This low cost approach will exploit and simplify the proven aerospace manufacturing techniques rather than rushing completely into the looser fabrication practices of the commercial boiler-tank plant It will recommend methods of decreasing cost which may involve some decrease in performance without compromising safety or dependability

Figure I-1 shows the breakdown of the total vehicle recurring cost into investment and operational costs for the Saturn IB, the Saturn V, the Thor-Delta, and the Titan III-C The investment cost includes the cost for fabrication, assembly, and checkout of the flight hardware (engines and stages) including their factory acceptance tests It includes the maintenance cost for the fabrication facilities and the sustaining engineering associated with the fabrication of the flight hardware Since these costs are limited to recurring costs, the investment cost does not include the costs incurred for rate tooling and facilities It does, however, include NASA support charges for services that NASA associates with the fabrication of the Saturn vehicle hardware This includes such costs as the operation and maintenance of the fabrication facilities supplied to the contractor by NASA, the DOD inspectors assigned in the fabricators' plant, and the product improvement charges assessed against the hardware The operation cost includes the cost of transporting the hardware from factory to launch site, all launch operations and launch support, all propellants and supplies, all integration and mission analysis work, and all the cost of operating and maintaining the launch base services and facilities that NASA assesses against its vehicles The Titan launch and launch support operations costs are significantly higher than those normally published by the Air Force, because the normal Air Force costs do not include items included in NASA costs These Air Force costs were taken from figures supplied to the NASA-DOD, Aeronautics/Astronautics Coordination Board but even they do not include all the cost items covered in NASA costs

Several comments can be made about the cost relationships shown The Saturn V shows a larger percentage of investment cost than the Saturn IB NASA bookkeeping policy directed that all costs which are common to both the Saturn V and Saturn IB should be charged to the Saturn V This results in an inflation of the investment cost of the Saturn V and a deflation of the investment cost of the Saturn IB. Therefore, the real difference in percentage of investment cost between the Saturn IB and V is not known

The percentage of investment cost for the Thor-Delta is significantly less than the percentage in the Saturn program Among the factors causing this difference, the fact that a much larger number of Thor-Delta's have been produced is considered significant. The larger the number of units produced,

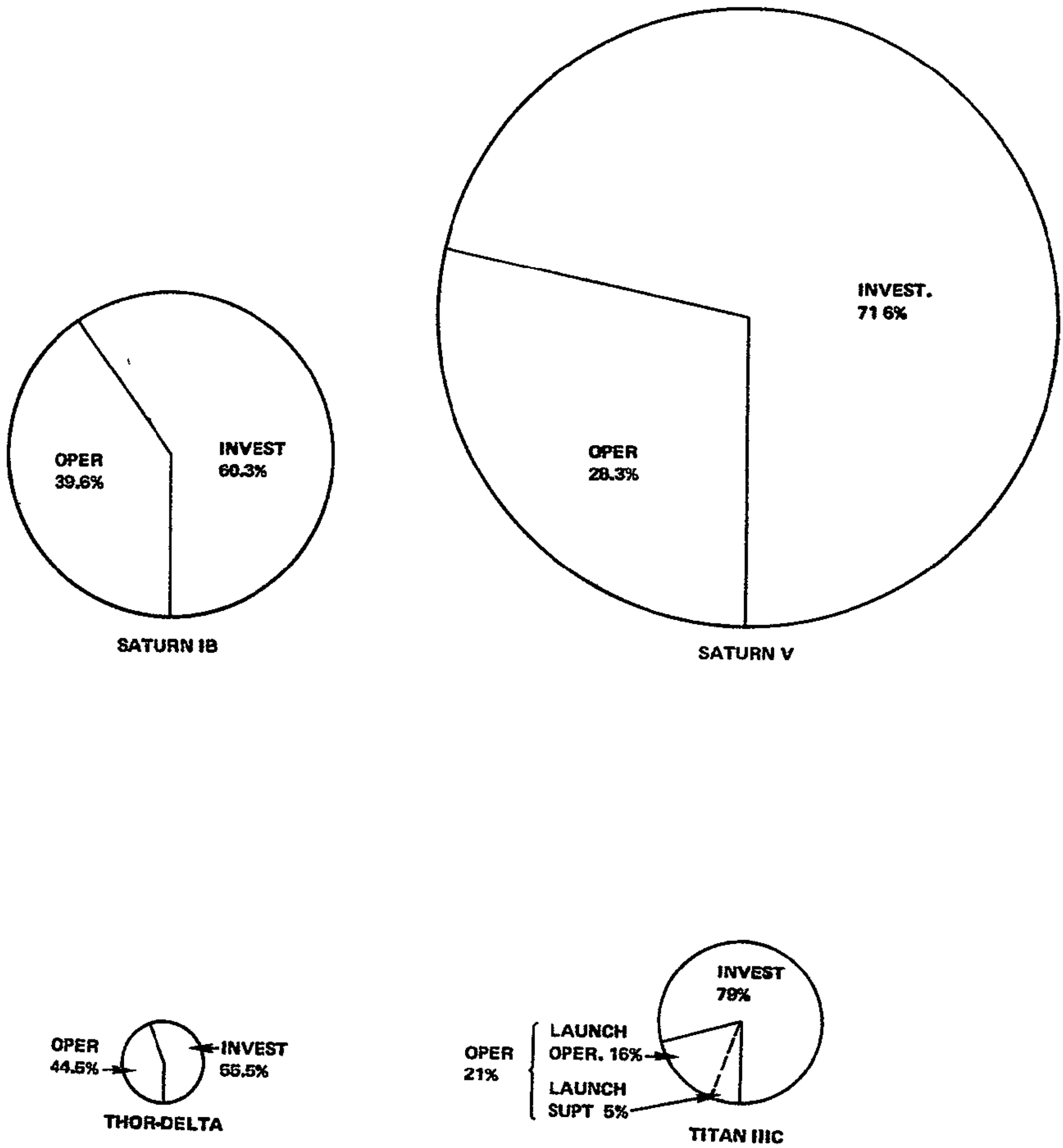


Figure I-1. Existing Expendable Launch Vehicles Investment and Operational Cost the farther they move down the learning curve and the less each hardware unit costs. While there is some learning associated with the operational costs, the charts suggest that operational costs do not decrease as rapidly as the hardware costs The actual conditions in the Thor-Delta program give credance to this conclusion The hardware unit production operations are largely repetitive and identical operations which permit maximum learning, however, the launch operations are effected by the payload configuration and mission assignment which tends to be different for each vehicle Also the very nature of the launch operations tends to encourage maintaining a stationary (level of effort) crew size which is less apt to be reduced in size than the fabrication crew No effort was made to adjust the costs of the two programs to remove the influence of the larger number of units produced on the Thor-Delta. This was not done for two reasons First, the purpose of this survey was not to manipulate the cost of the various programs until they agreed but rather to note the differences in the programs as they exist and by explaining why the cost relationships vary, gain insight into how these differences can be exploited to lower the costs in the new low-cost program Second, even though records are available, going back up a learning curve introduces so many other variables that the value of the resulting information for the purposes of this survey was not felt to justify the effort and assumptions necessary to obtain it.

If the only factor causing the difference in investment cost between the Saturn and the Thor-Delta program were the number of units produced, this would mean that the percentages were originally the same and the Saturn may be expected to arrive at the same ratio as the Thor-Delta. However, there are other factors involved which indicate there are program differences that are not totally dependent on the number of units produced and this decrease may not be assumed to be automatic The Thor-Delta program is managed by a small, well knit organization both in NASA and the contractor. Cost reductions are vigorously pursued. Incentives are applied in a manner that does not penalize future incentive to make additional changes. A single contractor is responsible for the vehicle Another factor is discussed under the functional breakdown of costs All these factors have contributed to decreasing both operational and investment costs However, investment costs appear to decrease faster than operational costs.

Turning to the Titan vehicle, the percentages of the costs are misleading. The Titan program is controlled and accounted differently than the NASA programs Cost elements included in NASA costs are not included in Air Force costs. The operational costs shown are broken into launch costs and support costs to give some indication of why the costs shown in these figures are larger than those normally published for the Titan. The background data shows $3 8 million as the cost of launching the Titan III-C The figure usually quoted by the Air Force is closer to $2 million Obviously it does not include some of the cost elements included by the joint NASA-DOD board

Figure I-2 shows the breakdown of the total vehicle hardware (investment cost) into stage and engine costs. The comparison of the Saturn vehicles shows that the instrumentation and guidance unit can easily be a larger part of the cost of a small vehicle than a larger one The IU is about half as large a part of the Saturn vehicles The Titan III-C shows the transtage as a sizeable portion of the vehicle cost but it includes propulsion and structures as well as instrumentation and guidance.

Figure I-3 shows the functional breakdown of the stage cost The data for a Titan stage is not available The data for the S-IVB and Delta stage shows a significant difference between the relationship of manufacturing cost to the quality control cost of the two vehicles On the Saturn program, the quality control effort is one-third as expensive as the hardware manufacturing effort On the Thor-Delta program, the quality control effort is one-fourth as expensive as the hardware manufacturing effort. Since the performance record of the Thor-Delta launch vehicle has demonstrated it is one of the most reliable and dependable vehicles in the nation's launch vehicle inventory, these figures indicate the quality control efforts can be investigated for methods of decreasing its costs and that a decrease in its cost can be accomplished in some programs without sacrificing safety or reliability Significantly enough, there are specific examples in the Thor-Delta program where inspection requirements have been designated as extreme and have been relaxed, over the objection of QC personnel, without any resulting degradation of quality. Similarly, there have been other areas where inspection requirements have been selectively tightened It appears likely that this responsiveness and flexibility is derived from the short, direct lines of communication existing between the small number of NASA and contractor personnel managing the program. Another factor may be sighted as a source of the difference in the production-QC cost relationship between the programs. The

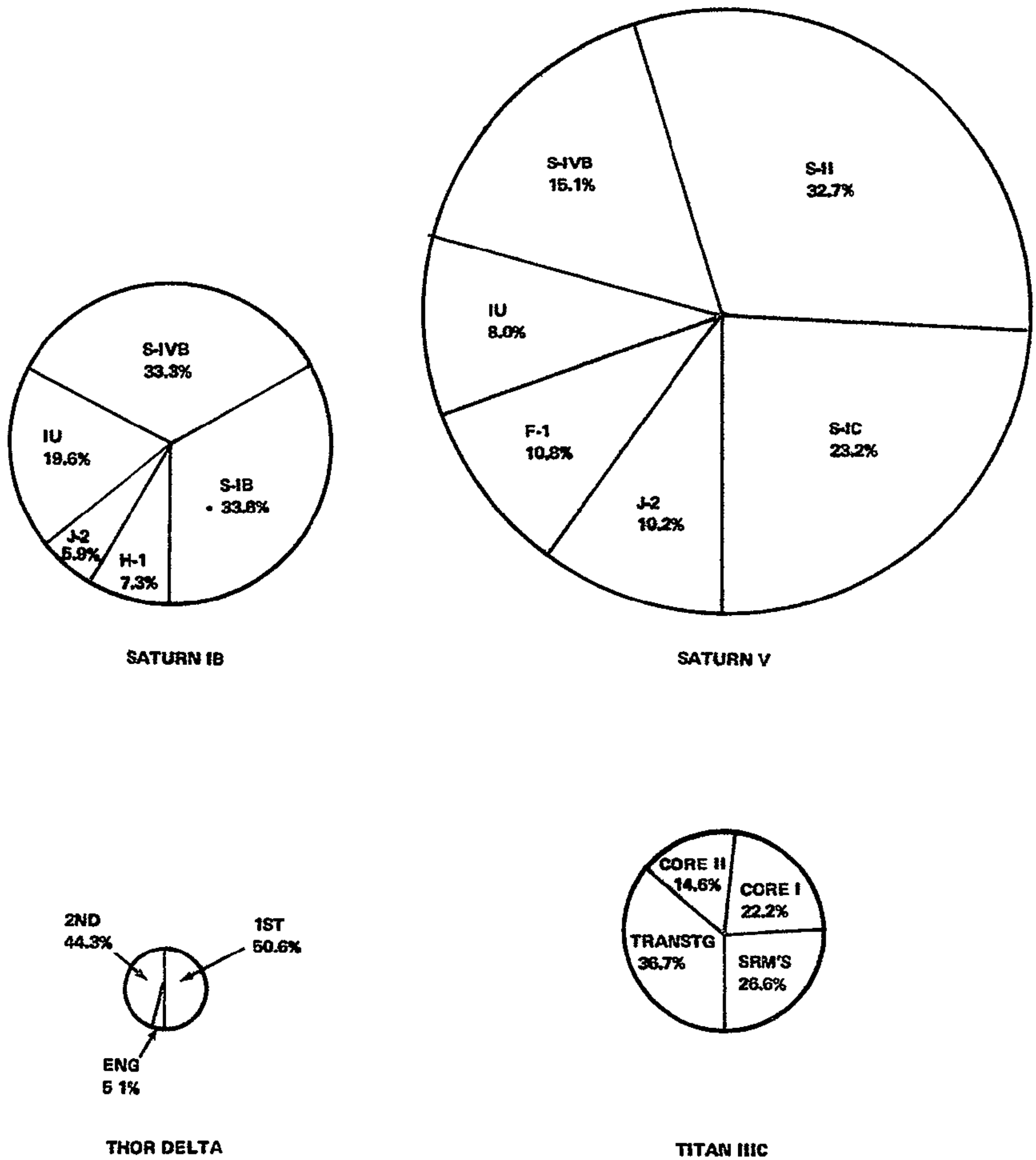


Figure I-2. Existing Expendable Launch Vehicles Investment by Stage

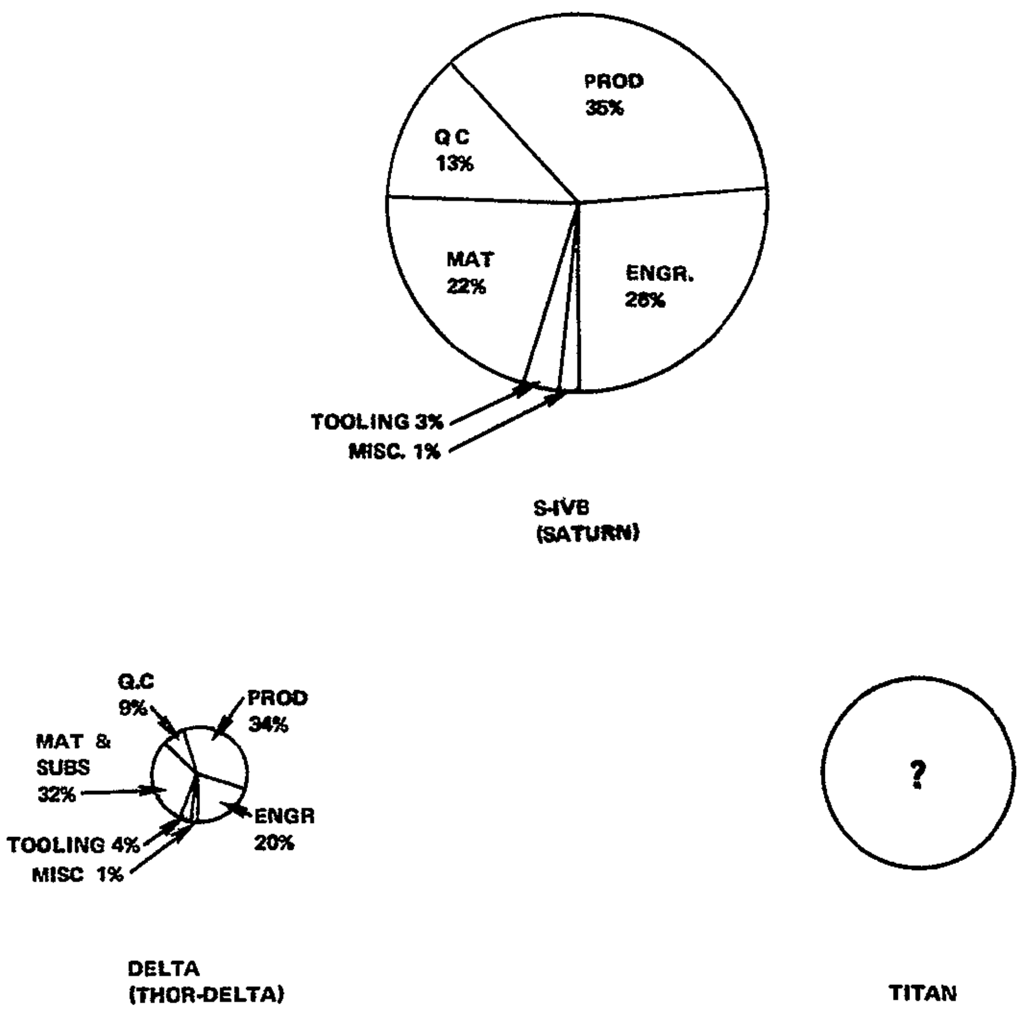


Figure I-3. Existing Expendable Launch Vehicles Investment by Function

Saturn is a manned vehicle and the Thor-Delta is an unmanned The definition of the difference between a manned and unmanned vehicle is hazy at some points but it is generally accepted that a manned program has special equipment to protect the crew such as a failure detection system If these added systems involved only additional inspection procedures, a manned vehicle would increase the ratio of inspection costs to fabrication costs However the added systems require additional fabrication costs as well It is therefore felt that the manned-unmanned status probably has a minimal impact on the proportion of the inspection to production cost in the two programs There is no indication that the inspection operations on the Delta are any less strict or detailed than that on the Saturn S-IVB (both of them are fabricated by McDonnell Douglas) simply because the Delta is unmanned system

Figure I-4 shows the percentages of the cost of selected subsystems associated with a stage of each of the different vehicles On the NASA programs, the structures portion is about 27% of the hardware costs. This suggests that structure is not as fruitful an area for cost reduction as some other areas such as electrical/electronics which accounts for a significantly larger portion of the stage cost The Titan seems to have exploited this potential area, since even though its structure is comparable in cost (dollar cost) to the Saturn structure, the ratio between structure and electrical/electronic systems cost is markedly lower. The simplification of the electronics on the Titan in comparison to the Saturn is evidenced by the smaller number of telemetry measurements on the Titan than on the Saturn. This appears to be at least partially derived from a difference in operating philosophy employed by the Air Force and the operating philosophy employed by NASA rather than completely inherent in the differences in system hardware

Figure I-5 shows the Saturn V breakdown of the costs paid to the stage and engine hardware contractors for costs incurred up to delivery of the completed and tested unit at the fabricators' plant. The broken line "Total Cost of Hardware" shows the height of the hardware bars if they were placed on top of each other The bar on the right side of the chart shows, to the same scale, the additional costs incurred by NASA and other NASA hired contractors for program support and services other than hardware procurement These costs include launch operations, base services, propellants, utilities, mission analysis, data

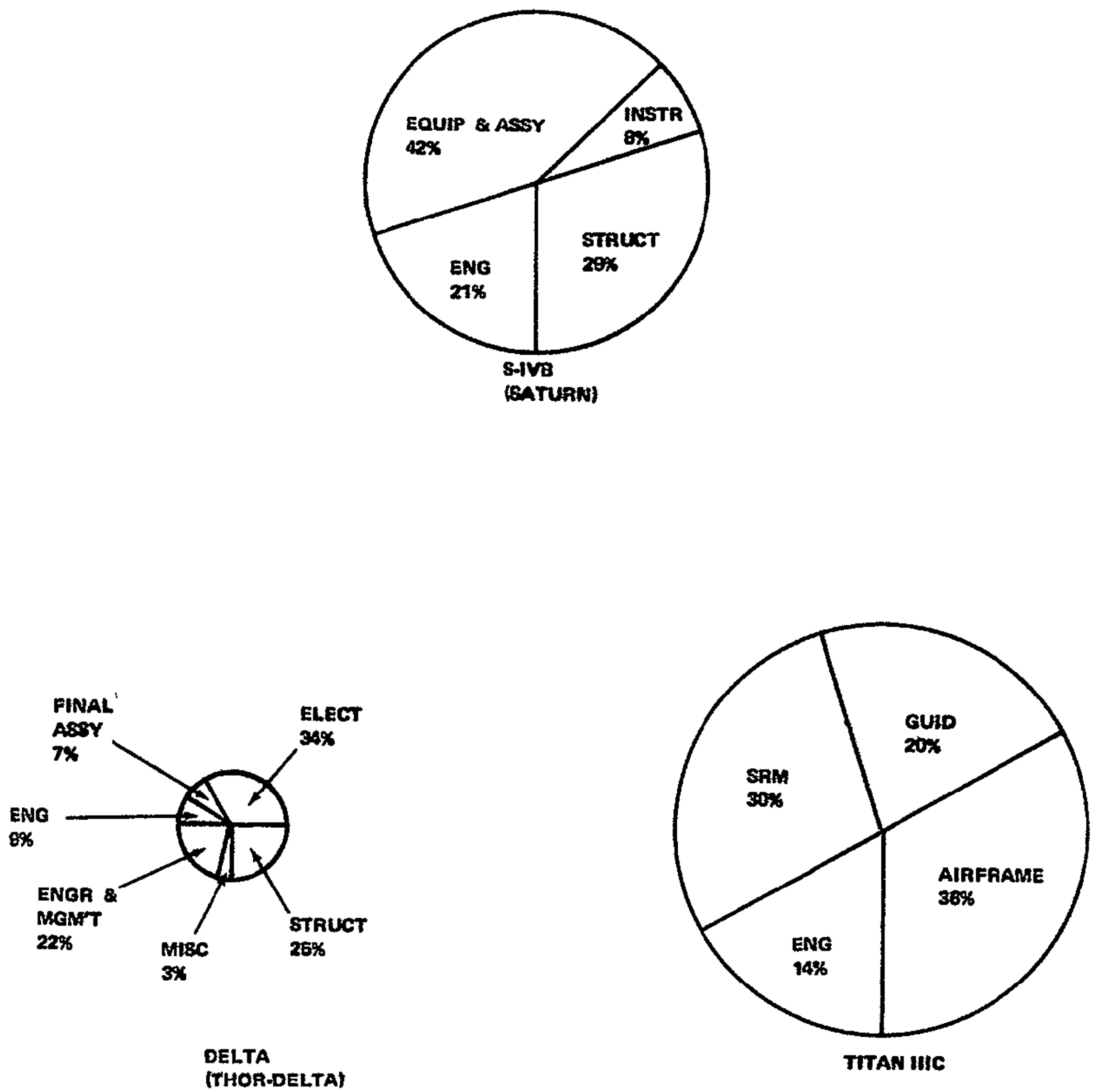


Figure I-4. Existing Expendable Launch Vehicles Investment by Subsystem

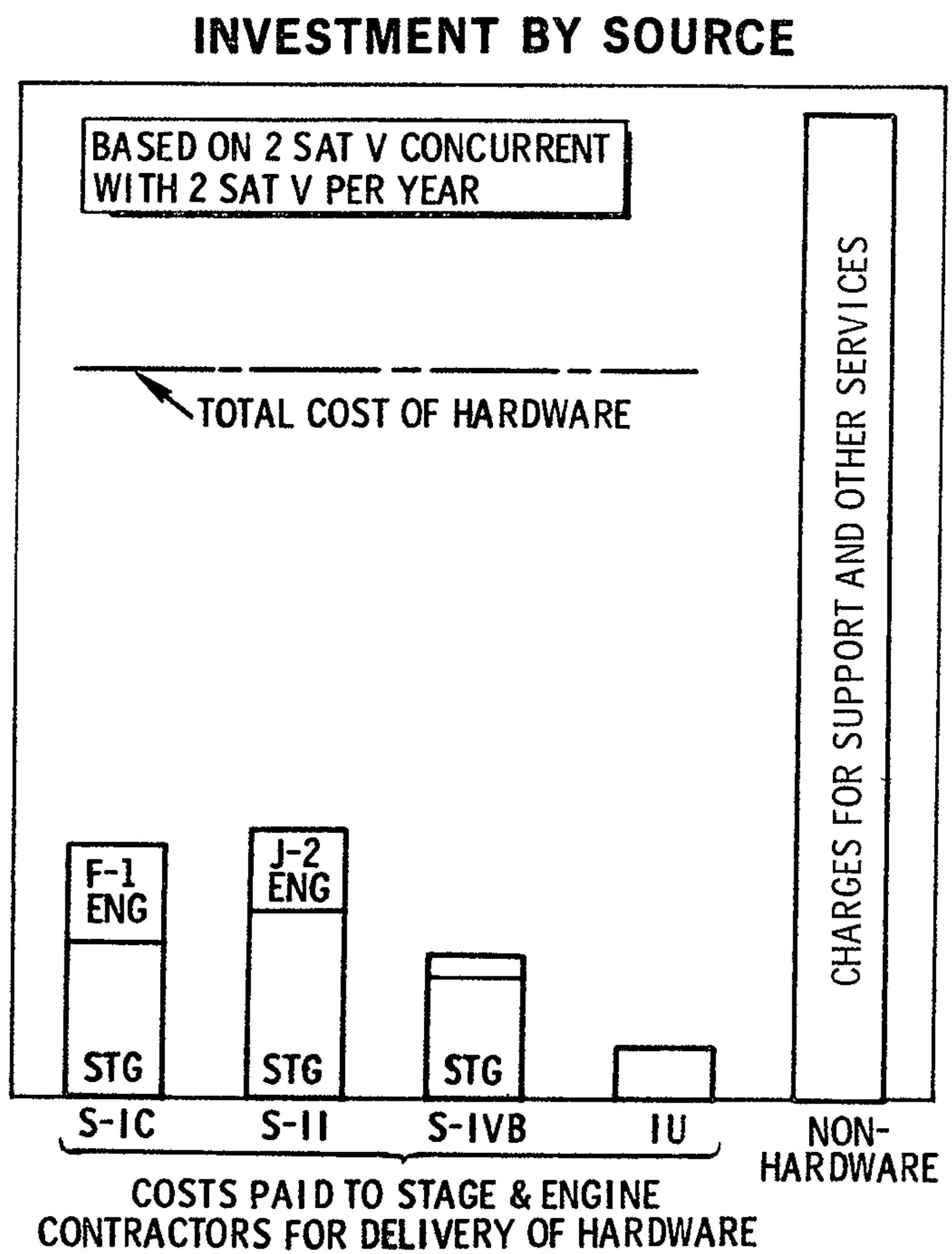


Figure I-5. Existing Expendable Launch Vehicles reduction, spares, product improvement etc They do not include the cost of NASA civil service personnel (AO funds) but include almost all other non-fabrication costs. As shown in Figure 1-5 these non-fabrication costs exceed the total hardware contractors' costs by a significant margin The hardware contractors' costs themselves include a significant amount of non-fabrication costs If the costs, such as sustaining engineering, documentation, program support and other non-fabrication costs, were separated from hardware costs and only fabrication and assembly shown as hardware costs, then the total non-fabrication charges would be more than three times the hardware costs This means that even if the hardware fabrication and assembly were done at contractor expense and given free to NASA, only one-fourth of the desired order of magnitude cost reduction would be achieved This can only mean then that non-hardware costs must be drastically reduced if the desired cost reduction goals are to be achieved Paper work, documentation, procedures, and services are the areas that must be changed. The Air Force management system differs from the NASA system in some of these areas and that is one reason the Titan costs are reported as less than NASA program costs

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ACKNOWLEDGEMENTS

The following were Principal Contributors to Volume II

M Beach
C W Cox
D L Crosby
R B Dobson
A J Goodwin
L J Pritchett
J R Quartucy
P Rosati
R A Sturdevant
J H Young